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The Journal of
THE ROYAL AERONAUTICAL SOCIETY

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MAY 1948 NO 449 VOL 52

THE COURAGE TO THINK THE CAPACITY TO DO

BIG WINGS AND LITTLE WINGS

The big wings of a
Lincoln bomber,
the little 'wings' of an
Armstrong Siddeley
propeller turbine engine —
both have their problems.
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THE ROYAL AERONAUTICAL SOCIETY
4 HAMILTON PLACE, PICCADILLY, LONDON, W.1

MAY NOTICES 1948

JOURNAL PREMIUM AWARDS

The Council have set aside an annual sum of £250 for the award of premiums for papers published in the *Journal* and the Council hope that members (or non-members) will contribute papers on their own special subjects.

CONTENTS OF MAY JOURNAL

Flight Testing of Helicopters, by W. Stewart, B.Sc.
Equilibrium Running of the Simple Jet Turbine Engine, by A. W. Morley, Ph.D., A.F.R.Ae.S.
Flight in Nature and in Human Society, by M. J. B. Davy, F.R.Ae.S.

WILBUR WRIGHT MEMORIAL LECTURE

The 36th Wilbur Wright Memorial Lecture, to be read on Thursday, 27th May 1948, by A. Gouge, B.Sc., F.R.Ae.S., will be: Size in Transport: A review of the development of the subject, with particular respect to size and the application of its lessons to Air Transport. The lecture will be held at 6 p.m. in the Lecture Hall at the Institution of Civil Engineers, Great George Street, S.W.1. Tea will be served at 5.30 p.m.

THE BUSK STUDENTSHIP IN AERONAUTICS

Application forms for the Busk Studentship in Aeronautics may be obtained from the Professor of Aeronautical Engineering, Engineering Laboratory, Cambridge, to whom they must be returned by the *1st June* 1948.

This Studentship was established in memory of Edward Teshmaker Busk who lost his life in 1914 while flying an experimental aeroplane.

The Studentship is of the value of about £165, tenable for one year from the 1st October. It is open to any man or woman who is a British subject and of British descent who has not attained the age of 25 years on 1st October 1948.

The object of the Studentship is to enable the holder to engage in research in aeronautics of the type in which Edward Busk was specially interested, that is of the type involving experimental as well as mathematical investigation.

ASSOCIATE FELLOWSHIP EXAMINATION

The next Associate Fellowship Examination under the present syllabus will be held on 1st, 2nd and 3rd June 1948. Those candidates who have registered with the Society will be informed individually of the detailed arrangements.

LECTURE PROGRAMME—1948

GLASGOW BRANCH

Saturday, 8th May 1948—Visit to Prestwick Airport. At 3 p.m.

Tuesday, 8th June 1948—Visit to The Glasgow Herald Office.

Saturday, 3rd July 1948—Visit to The Blackburn Aircraft Ltd., Clyde Factory, Dumbarton.

Thursday, 26th August 1948—" Lecturettes." At The Grand Hotel, Charing Cross, Glasgow.

Tuesday, 28th September 1948—A Film Evening, at 7.30 p.m., at Prestwick Airport.

Thursday, 28th October 1948—Lecture by a Branch Member, at 7.30 p.m., at The Grand Hotel, Charing Cross, Glasgow.

Tuesday, 30th November 1948—"Lecturettes," at 7.30 p.m., at Prestwick Airport.

Thursday, 23rd December 1948—ANNUAL GENERAL MEETING, at The Grand Hotel, Charing Cross, Glasgow.

Members are requested to notify the Secretary at least one week before any of the visits, if they intend to be present.

Visits are confined to members only, but visitors are welcomed at all Lectures, Lecturettes and Film Shows.

HATFIELD BRANCH

Wednesday, 19th May 1948—Film—Atomic Physics.
At 6.15 p.m. in the Lecture Theatre of the de Havilland Servicing School.

Wednesday, 16th June 1948—The Development of the Brabazon I, by G. P. Hepden, B.Sc., A.C.G.I., D.I.C.
At 6.15 p.m. in the de Havilland Senior Staff Mess.

Wednesday, 14th July 1948—Selection of Aeronautical Films.
At 6.15 p.m. in the Lecture Theatre of the de Havilland Servicing School.

Wednesday, 13th October 1948—Aircraft Photography, by John Yoxall.
At 6.15 p.m. in the de Havilland Senior Staff Mess.

Wednesday, 17th November 1948—Design Problems of Large Flying Boats, by H. Knowler, F.R.Ae.S.
At 6.15 p.m. in the de Havilland Senior Staff Mess.

Wednesday, 12th January 1949—Blind Landing Technique, by J. W. F. Mercer, B.Sc., A.C.G.I.
At 6.15 p.m. in the de Havilland Senior Staff Mess.

Wednesday, 16th February 1949—The Probable Role and Influence of Aircraft in Future Warfare, by Air Marshal Sir Robert Saundby, K.B.E., C.B., M.C., D.F.C., A.F.C.
At 6.15 p.m. in the de Havilland Senior Staff Mess.

Wednesday, 16th March 1949—Annual General Meeting.

LUTON BRANCH

Wednesday, 5th May 1948—Large Flying Boats, by H. Knowler, F.R.Ae.S., at the George Hotel, Luton, at 7 p.m.

GRADUTES' AND STUDENTS' SECTION

Tuesday, 11th May 1948—The Light Aeroplane and the Future of Private Flying, by P. G. Masefield, M.A., F.R.Ae.S., Director General of Long Term Planning and Projects, Ministry of Civil Aviation.

The meeting will be held in the Library of the Royal Aeronautical Society, 4 Hamilton Place, W.1, at 7.30 p.m. *The Library will not be open before 7 p.m.*

JOURNAL BINDING

Because of increased costs, the prices of binding of Journals will be as follows:—
1947 Volume 15/-.
Previous Volumes 17/-.

Journals should be sent direct to The Lewes Press, Friars Walk, Lewes, Sussex, and the remittance to the Secretary at the office of the Society.

CHANGES OF ADDRESS

To assist in keeping the records of members correct and up to date the Secretary will be glad if all members will notify him as soon as possible of changes of address.

When notifying changes please give the following particulars:—
Name (in block letters).
Grade of membership.
New address (in block letters).
Old address.

Changes of address must be received *before the 15th of the month* in order to be effective for the Journal for the following month.

NEW MEMBERS

Associate Fellows

Archibald Morton Ballantyne, Augustus Alfred Ward Barron, William Edward Breacker (from Graduate), Reginald Bruce Godfree, Zygmunt Stanislaw Ipohorski-Lenkiewicz (from Associate), Reginald Watson Kenworthy, John Reginald Lupton (from Graduate), Coorg Scrinivasachar Rangaswamy, Walter Percival Cuthbert Soper (from Graduate), Clifford Arthur Bowman Wilcock.

Associates

Peter Crossland, Thomas Ellison Dunwoody, Arthur Henry Nelson Emden (from Companion), Horace Joseph Alexander Galaud, Richard John Ifield, Arthur Kingsnorth (from Student), John Leworthy, Henry Lightly, Gordon McLachlan, Alfred John Macey, Herbert Kitchener Newton, Arthur Henry Porter, Ezz El Din Ramzi, Robert James Ritchie, John Ernest Dudley Scott, John Joseph Sullivan, Reginald Tillyard.

Graduates

Royston Herbert Blyth, Frederick Alfred Cousins, Edwin Jones, James Roland Miller (from Student), Raymond Matthew Ryan, Priykant Avantilal Vasavada.

Students

Students

John Ian Golesworthy, Merlin Kilick, Thomas Ambrose O'Sullivan, Stephen William Tonkin, Geoffrey Walter Webber.

Owing to an error in the Monthly Notices for April 1948, Sidney John Goates was shown as having been elected an Associate Fellow. Mr. Goates' name should have appeared in the list of recently elected Associates.

ACKNOWLEDGMENTS

The Council acknowledge with grateful thanks the gift of a collection of specially bound technical and historical books in English, French and German from Lord Nathan, Minister of Civil Aviation.

The Council also acknowledge with thanks a copy of "Aircraft Propeller Design" by F. T. Meacock from Miss A. D. Betts; books and periodicals from J. W. Bateman, Esq., Associate; and back numbers of the Journal from A. B. Carr, Associate Fellow.

ADDITIONS TO THE LIBRARY

Pamphlets in italics with location reference following in brackets. Books marked * or ** may not be taken out on loan.

N.A.C.A. Technical Reports

785—*Preknock vibrations in a spark-ignition engine cylinder as revealed by high-speed photography. Cearcy D. Miller and W. O. Logan.*

784—*Intercooler cooling-air weight flow and pressure drop for minimum drag loss. J. George Reuter and M. F. Valerino.*

754—*Operating temperatures of a sodium-cooled exhaust valve as measured by a thermocouple. J. C. Sanders, H. D. Wilsted and B. Mulcahy.*

786—*Performance of blowdown turbine driven by exhaust gas of nine-cylinder radial engine. L. R. Turner and L. G. Desmon.*

770—*Jet-boundary corrections for reflection-plane models in rectangular wind tunnels. R. S. Swanson and T. A. Toll.*

767—*The problem of longitudinal stability and control at high speeds. M. J. Hood and J. H. Allen.*

779—*The effect of increased cooling surface on performance of aircraft-engine cylinders as shown by tests of the NACA cylinder. O. W. Schey, V. G. Rollin and H. H. Ellenbrock.*

763—*Tests of airfoils designed to delay the compressibility burble. John Stack.*

764—*A method for calculating heat transfer in the laminar flow region of bodies. H. J. Allen and B. C. Look.*

760—*A method for estimating the knock rating of hydrocarbon fuel blends. N. D. Sanders.*

762—*A theoretical investigation of the lateral oscillations of an airplane with free rudder, with special reference to the effect of friction. H. Greenberg and L. Sternfield.*

755—*Requirements for satisfactory flying qualities of airplanes. R. R. Gilruth.*

757—*The measurement of fuel-air ratio by analysis of the oxidised exhaust gas. H. G. Gerrish and J. L. Meem, Jr.*

753—*Methods used in the NACA tank for the investigation of the longitudinal-stability characteristics of models of flying boats. R. E. Olsen and N. S. Land.*

761—*Identification of knock in NACA high-speed photographs of combustion in a spark-ignition engine. O. D. Miller and H. L. Olsen.*

771—*Review of flight tests of NACA C and D cowlings on the XP-42 airplane. J. Ford Johnston.*

787—*A theoretical investigation of the rolling oscillations of an airplane with ailerons free. Doris Cohen.*

N.A.C.A. Technical Memoranda

1106—*A ram-jet engine for fighters. E. Sanger and I. Bredt.*

1116—*Stability of plates and shells beyond the proportional limit. A. A. Ilyushin.*

1186—*The elasto-plastic stability of plates. A. A. Ilyushin.*

1185—*Systematic investigations of the influence of the shape of the profile upon the position of the transition point. K. Bussman and A. Ulrich.*

1175—*Fundamental aerodynamic investigations for development of arrow-stabilised projectiles. Hermann Kurzweg.*

N.A.C.A. Technical Notes

1366—*Effect of changing manifold pressure, exhaust pressure, and valve timing on the air capacity and output of a four-stroke engine operated with inlet valves of various diameters and lifts. J. C. Livengood and J. V. D. Eppes.*

1433—*Instability of outstanding flanges simply supported at one edge and reinforced by bulbs at other edge. S. Goodman and E. Boyd.*

1428—*Notes and tables for use in the analysis of supersonic flow. The Staff of the Ames I—by 3-foot Supersonic Wind-tunnel Section.*

1480—*Determination of plate compressive strengths. G. J. Heimerl.*

1440—*Further investigation of preheating and postheating in spot-welding 0.040-inch Alclad 24S-T. W. F. Hess and F. J. Winsor.*

1484—*Theoretical lift distribution and upwash velocities for thin wings at supersonic speeds. J. C. Evvard and J. R. Turner.*

1434—*A method of calculating the heat required for windshield thermal ice prevention based on extensive flight tests in natural icing conditions. A. R. Jones, G. H. Holdaway and C. P. Steinmetz.*

1471—*Experimental investigation of velocity distributions downstream of single duct glands. J. R. Wenke.*

1472—*The calculation of the heat required for wing thermal ice prevention in specified conditions. C. B. Neel, Jr., N. R. Bergrun, D. Jukoff and B. A. Schlaff.*

1494—*A method for estimating heat requirements for ice prevention on gas-heated hollow propeller blades. V. H. Gray and R. G. Campbell.*

1497—*A method of cycle analysis for aircraft gas-turbine power plants driving propellers. R. E. English and C. H. Hauser.*

1504—*Sideslip angles and vertical tail loads developed by periodic control deflections. Harvard Lomax.*

1509—*Cylinder-temperature and cooling-air-pressure instrumentation for air-cooled-engine cooling investigations. M. F. Valerino and S. J. Kaufman.*

1410—*An improved photoelastic method for determining plane stresses. C. B. Norris and A. W. Voss.*

1259—*A graphical method for interpolation of hydrodynamic characteristics of specific flying boats from collapsed results of general tests of flying-boat hull models. F. W. S. Locke.*

*865—*The direct-stress fatigue strength of 17S-T aluminium alloy throughout the range from 1/2 to 500,000,000 cycles of stress. E. C. Hartmann and G. W. Stickley.*

*864—*The electrical structure of thunderstorms. E. J. Workman, R. E. Holzer and G. T. Pelsor.*

*863—*Results of landing tests of various airplanes. J. A. Hootman and A. R. Jones.*

*862—*An investigation of the effectiveness of stiffeners on shear-resistant plate-girder webs. R. L. Moore.*

*861—*The effect of valve cooling upon maximum permissible engine output as limited by knock. Maurice Munger, H. S. Wilsted and B. A. Mulcahy.*

*850—*A preliminary investigation of the electrical structure of thunderstorms. E. J. Workman and R. E. Holzer.*

*849—*Normal-pressure tests of rectangular plates. Walter Ramberg, A. E. McPherson and Samuel Levy.*

*848—*Normal-pressure tests of circular plates with clamped edges. A. E. McPherson, W. Ramberg and Samuel Levy.*

*847—*Square plate with clamped edges under normal pressure producing large deflections. Samuel Levy.*

*846—*Bending of rectangular plates with large deflections. S. Levy.*

*935—*Dynamics of the inlet system of a four-stroke engine. R. H. Boden and Harry Schecter.*

*934—*Numerical procedures for the calculation of the stresses in monocoques. I.—Diffusion of tensile stringer loads in reinforced panels. N. J. Hoff, R. S. Levy and J. Kempner.*

*933—*Applicability of similarity principles to structural models. J. N. Goodier and W. T. Thomson.*

*932—*The numerical solution of compressible fluid flow problems. Howard W. Emmons.*

*931—*Guides for preventing buckling in axial fatigue tests of thin sheet-metal specimens. W. C. Brueggeman and M. Mayer, Jr.*

*886—*Torsional elastic properties of 18:8 chromium-nickel steel as affected by plastic deformation and by heat treatment. R. W. Mebs and D. J. McAdam, Jr.*

*887—*A portable calibrator for dynamic strain gages. A. B. McPherson.*

*888—*Torsion of flanged members with cross sections restrained against warping. H. N. Hill.*

*889—*Effects of prior fatigue-stressing on the impact resistance of chromium-molybdenum aircraft steel. J. A. Kies and W. Holshouser.*

*890—*Pressure distribution on the fuselage of a midwing airplane model at high speeds. James B. Delano.*

*805—*Effect of aging on mechanical properties of aluminium-alloy rivets. F. C. Roop.*

*804—*Wind-tunnel investigation of the effect of vertical position of the wing on the side flow in the region of the vertical tail. Isidore G. Recant and A. B. Wallace.*

*803—*Some effects of rainfall on flight of airplanes and on instrument indications.* *R. V. Rhode.*

*802—*Tests of a gust-alleviating wing in the gust tunnel.* *C. C. Shufflebarger.*

*801—*Tests of round and flat spoilers on a tapered wing in the NACA 19-foot pressure wind-tunnel.* *C. J. Weinzinger and J. S. Bowen.*

*835—*Analysis of ground effect on the lifting airscrew.* *Montgomery Knight and R. A. Hefner.*

*834—*The calculation of span load distributions on swept-back wings.* *William Mutterperl.*

*833—*Ultimate stresses developed by 24S-T sheet in incomplete diagonal tension.* *Paul Kuhn.*

*832—*Relief of residual stress in streamline tie rods by heat treatment.* *R. E. Pollard and F. M. Reinhardt.*

*831—*A method of determining the equilibrium performance and the stability of an engine equipped with an exhaust turbosupercharger.* *James Buchanan Rea.*

*761—*Pressure-distribution investigation of an NACA 0009 airfoil with an 80 per cent. chord plain flap and three tabs.* *M. B. Ames, Jr., and R. I. Sears.*

*762—*The flow of a compressible fluid past a sphere.* *Carl Kaplan.*

*763—*Wind-tunnel investigation of two airfoils with 25 per cent. chord wing and plain flaps.* *M. B. Ames, Jr.*

*764—*Flight investigation of control-stick vibration of the YG-1B autogiro.* *F. J. Bailey, Jr.*

*765—*A method of measuring piston temperatures.* *B. Pinkel and E. J. Manganiello.*

*766—*A study of the air movements in two aircraft-engine cylinders.* *D. W. Lee.*

*767—*The effects of engine speed and mixture temperature on the knocking characteristics of several fuels.* *Dana W. Lee.*

*768—*Correlation of knocking characteristics of fuels in an engine having a hemispherical combustion chamber.* *A. M. Rothrock and A. Biermann.*

*769—*Spin tests of a low-wing monoplane in flight and in the free-spinning wind tunnel.* *O. Seidman and W. H. McAvoy.*

*770—*The lateral stability of equal-flanged aluminium-alloy. I.—Beams subjected to pure bending.* *C. Dumont and H. N. Hill.*

A.R.C. Reports and Memoranda

2192—*Investigation of induction swirl on single-cylinder sleeve-valve engines.* *Ricardo & Co.*

2202—*Elevator control on Firefly F. Mk. I and NF. Mk. II aircraft.* *S/Ldr. D. R. H. Dickenson.*

2206—*The pressure distribution on some flat laminar aerofoils at incidence at supersonic speeds.* *G. N. Ward.*

2193—*Data for the design of smooth wing surfaces.* *D. M. A. Leggett.*

2199—*Review of remote indicating systems for aircraft instruments.* *F. Postlethwaite.*

2121—*Fuselage vertical bending elevator flutter on the Typhoon.* *H. A. Jahn, G. H. L. Bunton and I. T. Minhinnick.*

2191—*Flight measurements of static and dynamic longitudinal stability characteristics of two twin-engined monoplanes—the Beaufighter IF and the Boston III.* *D. E. Morris and H. G. Alston.*

2106—*An air-injection method of fixing transition from laminar to turbulent flow in boundary layer.* *A. Fage and R. F. Sargent.*

2179—*Effect of variation of airscrew tip speed and drag critical speeds at two forward speeds.* *R. C. Pankhurst.*

2177—*Free streamline suction slots.* *E. J. Watson.*

2154—*An elaboration of the criterion for wing torsional stiffness.* *A. R. Collar, E. G. Broadbent and Elizabeth B. Puttick.*

The effect of taper on wing flexure-torsion flutter speeds. *E. G. Broadbent and Elizabeth B. Puttick.*

1967—*The difference between the spinning of model and full-scale aircraft. G. E. Pringle.*
2228—*Tests of landing brake propeller on a fighter aircraft. A. J. Wilson.*
2214—*Free oscillations of an aerofoil about the half-chord axis at high incidences, and pitching moment derivatives for decaying oscillations. J. B. Bratt, K. C. Wright and A. Chinneck.*
2211—*The effect of brake flaps on an aerofoil at high speeds.*
Part I.—Measurements on the effect of brake flaps on an aerofoil at high speeds. A. E. Knowler and F. W. Pruden.
Part II.—Further wind-tunnel tests of brake flaps at high speeds. F. W. Pruden.
2208—*The measurements of structural stiffness of aircraft. E. G. Broadbent and D. L. Woodcock.*
2207—*Design of a fixed-pitch pusher propeller coupled to a free-running turbine. A. B. Haines.*
2203—*Lift characteristics for thin Clark Y propeller sections at low and negative angles of incidence. A. R. C. MacDougall.*

I.C.A.O. Montreal

4976—*Supplementary procedures for the South American Region.*
4977—*Supplementary procedures for the South Atlantic Region.*
4820—*Final report of Meteorological Division special session. September* 1947.

The following reports have also been received:—

British Intelligence Objectives Sub-Committee

B.I.O.S. Final Reports, Nos. 1697 and 1700.
B.I.O.S. Miscellaneous Report, No. 61.

Monografie Scientifiche di Aeronautica

Reports, Nos. 4, 6 and 7.

Publications Scientifiques et Techniques du Ministere de l'Air

Reports, Nos. 207 and 209.

J. LAURENCE PRITCHARD,
Secretary.

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The Royal Aeronautical Society

ACTIVITIES OF THE GRADUATES' AND STUDENTS' SECTION

May 1948

The Section's Annual General Meeting was held in the Council Room at the Society on Wednesday 21st April 1948. A report of this meeting and a list of the new Committee will appear in the next issue of the JOURNAL.

On 14th April, Mr. W. Tye, F.R.Ae.S., read a paper to the Section on "The Influence of Recent Civil Airworthiness Requirements on Civil Aircraft Design," and it is hoped to include a summary of this lecture in a later issue of these notes.

LECTURE

Summary of a Paper on Wing Testing by K. R. Obee, Grad. R.Ae.S., delivered to the Section on 15*th October* 1946.

The paper approaches the subject from the stress aspect and is divided into four sections corresponding to the normal test report, namely:—

1. Object of test.
2. Method—which is further sub-divided into preparation for the test and the actual testing process itself.
3. Results.
4. Conclusions.

1. *Object of Test.*

Primarily, it is intended as a strength check on the wing structure to ensure that the main spar or spars and torque-carrying structure have adequate strength under the loads imposed upon them.

It is a practical, as opposed to a theoretical, method of checking the strength of the wing. Thus the second object of the test is to check the stressing method used and, in the case of new types of construction, the object may even be to evolve stressing methods from *ad hoc* research.

A final most important object of the test is to find whether an increase in all-up weight above that allowed by the minimum reserve factor is justified.

2. *Method of Test.*

Preparation for the test includes calculation of loads and design of test gear. The paper deals with the two-spar skin-covered wing, but in general the remarks are applicable to other structures.

The test is on the main structure only, in which the theoretical loading over the wing is replaced by loading points chosen mainly along the line of the spars in such a way that the loads reaching the main spar and torque box represent the design conditions.

Three cases may be necessary to test the primary structure, namely, for the spar booms, spar webs or diagonals, and the skins, corresponding to maximum B.M., S.F. and torque respectively. Accuracy is governed by the number of spanwise loading points, the chordwise points being in a minimum of two lines to produce the correct torque. For a single spar wing, the two lines of loading points are usually chosen along the spar and leading edge. Having chosen the loading points, the loads to be applied are found by approximating to the design S.F. curve by a series of steps so drawn that the inter-section of the horizontal lines with the design curve correspond to the critical points where the maximum accuracy is desired.

If the S.F. test curve is carefully chosen the B.M. curves will be closely similar, the main difference being that the design case is a smooth curve and the test case a series of straight lines. The loads to be applied at the chordwise points are found by drawing a series of steps in the torque curve in a similar manner to the S.F. curve.

An additional check is often made of the strength of the leading edge attachments to (say) the front spar by applying loads along the leading edge, and in the case of a single spar structure under these conditions a third line of loads is required to produce the correct torque.

The standard method of applying loads on the spars is through a shackle fixed to a beam straddling the spar and attached by bolts passing through the skin to a similar beam on the opposite surface which transmits the load to the spar. At the leading edge, the load is applied through a strap passing round and riveted to it. Loads from the engine and undercarriage are applied through a superstructure attached to the pick-up points. Local strengthening of the structure at the load points is usually necessary, and test gear is stressed for a safety factor of three.

Allowance is made for loads due to the static weight of the structure and its test rig before starting the test. Positioning of strain gauges is of great importance as, although 2,000 or more readings are probably required to obtain the maximum information from a wing test for the four-motor 80,000 lb. A.U.W. category of aircraft, 500 must usually suffice and these must be placed so that they will help the discovery of such obvious unknowns as spar end loads, skin and spar web shears, the effect of large holes in the torque box, and the ability of the skin to carry loads due to wing bending. The test is performed in a boxwork steel structure, familiarly known as an Abbey, Cathedral, etc., according to its size, to the floor of which is anchored the wing. Loads are applied through a system of levers reaching to the roof where they are connected to several hydraulic screw jacks. Vernier tapes, dial and strain gauges are placed where required.

The wing is tested from tip to tip, and it is most important that the anchorage at the centre section should include at least a chord length of the fuselage fore and aft of the wing in order to ensure the accuracy of the centre section reactions.

As only one case can be taken to destruction this is usually the maximum B.M., the other two being taken to proof load. Repeated loadings are made, but so far the stage of fatigue testing large wings has not been reached.

3. *Results*

Load-deflection curves—both spanwise and as a twist in the wing—are produced from readings noted at each halt in the loading and these form the basis of checking or evolving new methods of stressing. The behaviour of the wing is also noted, photographically if possible, the first signs of skin wrinkles being an obvious example.

Strain gauge readings are recorded by a pen recorder on strips of paper, one for each loading. A limitation of the strain gauge is that it only measures strain, which is not so important as stress, and stress can only be deduced from strain so long as Hooke's Law holds.

4. *Conclusions.*

The results may confirm the theoretical calculations, but if not, the reasons must be investigated. Care must be taken not to draw wrong conclusions when the method of testing may be at fault. In conclusion, the paper emphasises that thrift in testing may often prove a false economy as the results obtained thereby may be inconclusive and misleading.

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FLIGHT (weekly) provides the most authentic information on world aeronautical affairs. It covers every aspect of development and progress in aircraft and power unit design and operation. AIRCRAFT PRODUCTION (monthly) is the journal of the aircraft manufacturing industry, specialising in tools and works production processes. The editorial staffs of each journal are experts in their own particular sphere, with unrivalled experience and resources. Both journals serve the interests of all concerned with the future progress of British Aviation. Technical information is supplemented by brilliant functional drawings. Circulation is world-wide. Annual subscriptions (Home and Overseas) FLIGHT £3.1.0, AIRCRAFT PRODUCTION £1.14.6.

Published in conjunction with these journals, FLIGHT HANDBOOK (212 pages, 7/6 net) is essentially a manual for the student, whilst GAS TURBINES AND JET PROPULSION (272 pages, 12/6 net) by G. Geoffrey Smith, has been widely adopted as the standard textbook on the subject by Universities, Technical Institutions and Training Centres everywhere.

ASSOCIATED ILIFFE PUBLICATIONS

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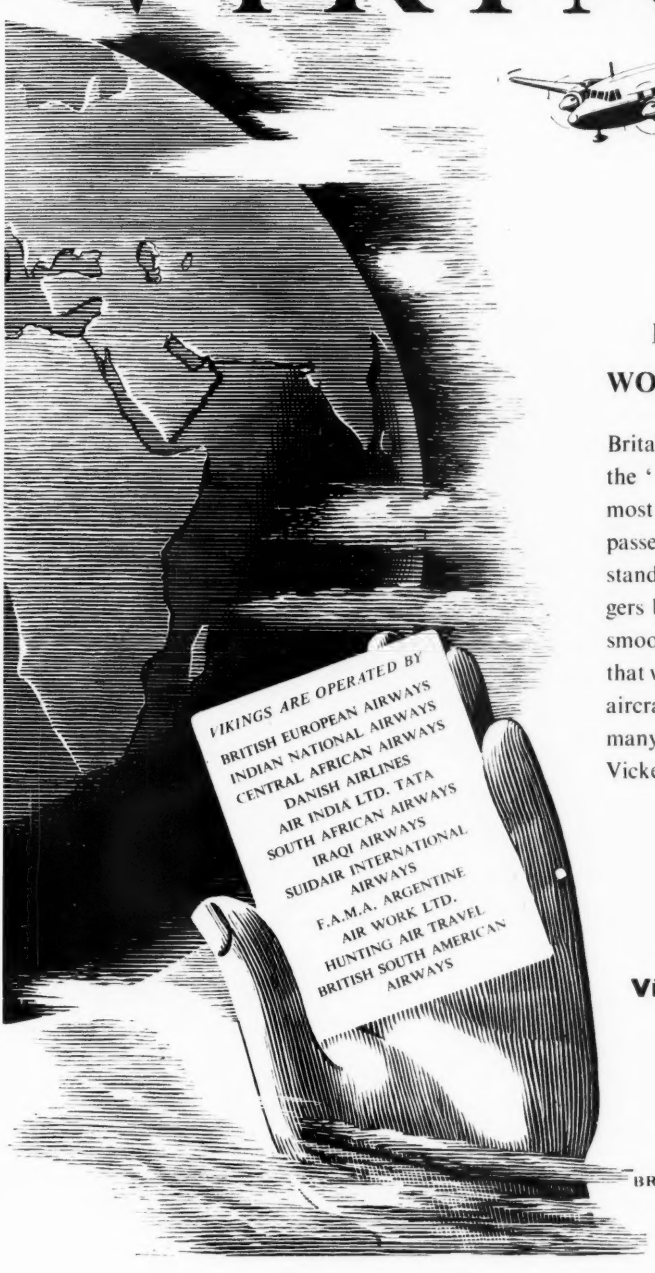

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THE ROYAL AERONAUTICAL SOCIETY

The 745th Lecture read before the Society—on the 29th January 1948 at the Institution of Civil Engineers, Great George Street, London, S.W.1. In the Chair, the President, Dr. H. Roxbee Cox, D.I.C., F.R.Ae.S.

FLIGHT TESTING OF HELICOPTERS

by

W. STEWART, B.Sc.

Mr. Stewart took his degree at Glasgow University in 1942, *specialising in mechanical engineering, aeronautics and advance mathematics. He joined the Aero Department, Flight Section, of the Royal Aircraft Establishment and worked on helicopters and is now head of the Helicopter Group which was formed at Farnborough in* 1945. *He is a member of the Aeronautical Research Council Committee on Helicopters.*

1. INTRODUCTION.

IT is only within recent years that the helicopter has emerged from the experimental aircraft category to a position in which it can be considered as an effective operational and commercial means of aerial transport. The development of the helicopter to its present stage is due to extensive work in many countries, particularly in America, and it is unfortunate, but unavoidable, that Britain was unable to take a larger share in this work.

In this country, considerable theoretical work on the aerodynamics of autogyro blades was done by Glauert and Lock.[1, 2] Glauert extended his theory of the autogyro principle to a power-driven rotor in vertical flight and then to horizontal flight, with the axis still vertical, the rotor being driven by small propellers at the blade tips. A later theoretical analysis by Squire allowed the rotor to tilt and produce a forward component of the thrust to overcome the drag of the rotor and fuselage.

On the practical side, following earlier work on the autogyro (particularly the developments due to Juan de la Cierva) and on the gyroplane, a lapse occurred in the development of rotary wing aircraft, although there were many advocates for this type of machine. This enthusiasm has recently been revived and is shown particularly in the helicopter activities of several firms in this country.

A small research group dealing with flight investigations on the helicopter was formed in the Aerodynamics Department at the Royal Aircraft Establishment at the end of 1945. It is mainly with the work of this group that I shall be dealing in this lecture.

2. DEVELOPMENT OF FLIGHT TESTING.

With the initiation of flight research into the aerodynamic characteristics of the helicopter, the first essential in each section of the work was to establish satisfactory testing techniques. The vast amount of experience with the fixed-wing aeroplane and in the earlier work on the rotary wing type was of great assistance in the rapid development of suitable procedures. Thus, it will be apparent in various sections of this lecture that the flight testing techniques adopted follow closely on fixed-wing aeroplane practice.

Nevertheless, the fundamental differences brought about by the inherent characteristics of the helicopter meant considerable modification to such existing techniques and the introduction of several new ones. Not only had the methods to be made applicable to the helicopter as a type of flying machine, but also to the specific type of helicopter. The Sikorsky R-4B, on which most of the flight testing has been done both in America and in this country, has several undesirable features from the point of view of its use as a research helicopter. The lack of power, which renders it unable to hover in free air at normal loading; the vibration in the pilot's control stick, which makes the measurement of control behaviour very difficult; and the "unaerodynamic" fuselage all produce serious difficulties to the pilot and to the research technician in flight testing this type of helicopter.

For the purpose of the present lecture, the recent flight testing work on helicopters may be divided into four sections:—

(a) The nature of the flow through the rotor disc.
(b) Performance testing.
(c) Stability and control.
(d) Blind flying.

The fundamental problem on the helicopter —whether the information is required for performance estimation or in stability and control work—is the nature of the flow through the rotor under its various operating conditions. Many years ago Lock made an investigation of this problem with models in the wind tunnel, but until the present research no work in flight had been attempted.

The sections on the development of performance testing and stability and control assessment follow quite closely on fixed-wing aeroplane practice. Thus, in this lecture, emphasis is given to the conditions where the differences caused by the helicopter's characteristics are important.

In helicopter flight testing, as in all forms of research work, the use of proper instrumentation is of major importance. The use of an automatic-observer was indispensable in many of the tests described in the following sections. The extensive use of instruments is essential if the pilot's impressions of the flying characteristics of the helicopter are to be understood and rationalised in scientific form. Several new instruments and techniques have already been used for the present flight testing work and further developments will be required for the many problems which still remain unsolved.

3. FLOW THROUGH THE DISC.

The nature of the flow through the rotor disc is of fundamental importance in many branches of helicopter work. For many years it was assumed that the induced velocity at the rotor disc was uniform, but departures of the actual distribution from this asssumption have a marked effect on the performance and stability and control characteristics. Recently, empirical factors based on general experience have been introduced to allow for these variations.

Flight testing in connection with this work is naturally of a specialised research nature, but in view of its fundamental influence on the understanding of the helicopter, it has received particular attention in recent flight investigations.

The rotor has three distinct working conditions, defined in general terms as:—

(a) propeller working state
(b) vortex ring
(c) windmill brake state.

The simplest way of thinking of these states is to start by considering a propeller moving in the direction of the thrust, *e.g.* a helicopter flying vertically upwards. This is the propeller working condition and the thrust is derived by increasing the momentum of the air as it passes through the disc. Propeller theory can be applied to this state, either in the simple form of an assumed constant induced velocity through the rotor disc or, by strip theory, where each annular area is treated independently. It can be shown that the power required for a given thrust is a minimum for the uniform distribution, but this cannot be obtained in practice. Obviously there must be a loss at the tip and a non-lifting section at the root. Even neglecting the root and tip sections, the required twist and/or taper of the remainder of the blade to give uniform distribution in a particular condition would produce adverse effects in other flight conditions.

The limiting condition for the propeller working state is the static thrust case or hovering. As the helicopter begins to descend it enters the vortex ring state, and is now moving in the opposite direction to that of the thrust vector. In this state, the rotor is still pushing air downwards but the normal propeller theory breaks down. The failure of the theory is easily realised, if an attempt to apply simple momentum theory is made. In this the relative free air is approaching the rotor from below and to provide sufficient momentum to support the helicopter, the required change in velocity is greater than that of the upward flow and the necessity of maintaining continuity of flow makes such a state impossible.

Beyond the vortex ring conditions exists the windmill brake state, in which the rotor derives its energy from the air. In this condition the air is flowing upward through the rotor disc, *i.e.* in the direction of the thrust vector, the thrust being produced by slowing the air down as it passes through the disc. Propeller theory again becomes applicable.

The boundary condition between the vortex ring and windmill brake states is the condition where the work done by the air is exactly that required to overcome the profile drag of the blades. From momentum considerations this would be the ideal condition where the helicopter could be supported by

reducing the velocity of the upflowing air to zero.

As the helicopter moves into forward flight, the same three working conditions are maintained. The relative air flow may be divided into two components, the axial velocity parallel to the rotor thrust and the tangential velocity at right angles. The boundaries between the various states are determined in a similar way to the vertical flight conditions from the nature of the axial component.

In forward flight, where the translational velocity is combined with the rotational velocity, the radial thrust distribution will vary with the azimuth position of the blade.

Wind tunnel tests on model propellers by Glauert and Lock established the flow conditions in the various working conditions, but there has always been some doubt on their applicability to full-scale rotor performance of the helicopter.

The flight tests described here were initiated to investigate these operating conditions for the full-scale helicopter. While it is comparatively easy to derive the existing conditions in general terms, the accurate measurement of the conditions presents many difficulties, mainly because of the very small order of the terms involved. Thus, for present helicopter disc loadings the increase in total head of the air passing through the rotor in hovering flight is of the order of $\frac{1}{2}$ inch of water and in forward flight the induced flow is considerably less than this. The small order of the induced velocities makes it essential to measure the flight conditions of the helicopter very accurately and to maintain these conditions absolutely steady during the test runs.

3.1. Hovering flight.

The obvious starting point for an investigation of the flow conditions through a helicopter disc is in hovering flight. In this condition the helicopter is stationary, the flow is directed downward through the rotor and the radial distribution is practically similar for any azimuth position of the blade.

It is well established that the energy imparted to the air by the rotor is partly kinetic and partly pressure energy. As the slipstream develops a contraction of the boundary takes place with the velocity increasing until normal atmospheric pressure has again been attained. It has been shown by Glauert and others that the velocity in the fully contracted slipstream is twice that at the rotor disc. Thus, measurements of the slipstream velocities, to be of any value, must be measured at the disc or in the fully contracted section downstream.

Attempts to measure the velocity close to the disc by the hot-wire anemometer method were made in America. Even for the closest practical position, the correct distribution is not obtained because of the rapidity with which the slipstream develops. In the present tests, the method of measuring the velocities downstream was used. In addition, velocity measurements were made at several distances below the disc to investigate the slipstream development.

The induced velocity through the rotor is very small and to obtain the required accuracy some form of magnification is necessary. The flight tests were made with a double venturi, in which the differential pressure between the total head pitot and the static pressure in the centre throat gave an amplification of 13 in terms of $\frac{1}{2}\rho V^2$.

The venturi instrument was carried on a pole projecting radially from the top of the port undercarriage leg (4.7 ft. below the flapping hinge plane). Measurements could be obtained for radial positions 0.28 to 1.1 of the blade radius. Provision for vertical adjustment of the instrument to cover a range of 4 ft. above and below the pole was also made.

A special technique of streamline visualisation by smoke filaments was developed to enable measurements at any point in the slipstream to be referred to the appropriate radial position at the rotor disc. It was found that the smoke streamer released from a generator continued through the disc without any serious dispersion of the smoke.

Hovering flight presents peculiar difficulties since the helicopter rotor must be at least 40 ft. above the ground to eliminate the ground cushion effect. It then becomes difficult to find a means of positioning the generator close to, and above, the rotor disc. The practical solution to this problem was to attach the smoke generator to the end of the jib of a 60 ft. crane and to hover the helicopter under this. Ground signals to the pilot were necessary to achieve the required helicopter positioning.

The development of the slipstream for hovering flight, as obtained from the smoke filaments, is shown diagrammatically in Fig. 1. The rapid contraction of the slipstream close to the rotor shows the

inapplicability of measurements close to the disc. The velocities measured at the fully developed section (XX in Fig. 1) and corrected to their appropriate positions on the disc are given in Fig. 2. The theoretical distribution, calculated by propeller strip theory, is also included in this figure for comparison. In the flight tests, the exact limit of the slipstream boundary is difficult to determine by the smoke filaments because of the influence of the tip vortex system. This makes it difficult to allow for the tip loss and the boundary has therefore been continued to the tip of the disc. Similarly, no allowance for tip loss was included in the theoretical treatment, so that the two curves can be compared directly. The agreement is well within the experimental accuracy of the tests and confirms the application of strip theory to the helicopter rotor.

Integration of the experimental and theoretical induced velocity gradings gave the same thrust. This thrust was about 10 per cent. greater than the weight of the helicopter. When an allowance for the fuselage vertical drag (estimated at 90 lb.) had been made, the remainder was considered as representing the tip loss. This value of the tip loss of 7 per cent. agrees with existing data on the subject. As the value was derived from the difference of two large quantities, it does not constitute a verification of tip loss estimation.

The comparison with strip theory was extended to power considerations, but measurement of the angles of swirl could not be made with great accuracy because of fluctuations in the slipstream and the small order of the required angle (about 6 degrees for the present conditions). Satisfactory agreement with theory was obtained within the experimental limitations of the slipstream swirl measurements.

3.2. Vortex ring state.

As the helicopter descends from hovering flight it enters the vortex ring state, a condition in which little is known of the physical construction and where no theory has, as yet, been devised which could be applied satisfactorily. The air associated with the rotor does not form a regular slipstream development but there exists a circulation of air in a very turbulent state about the rotor, the flow being directed downwards through the rotor and then moving radially outward and upwards outside the rotor disc. Some of this air as it passes upwards above the rotor is again drawn inwards and through the rotor.

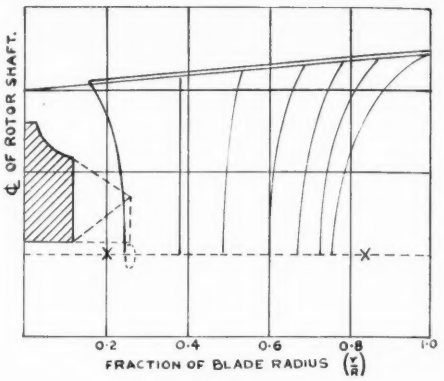


Fig. 1.
Slipstream pattern in hovering.

Thus, some of the air may circulate through the rotor many times, in the manner from which this state derives its title, namely, the vortex ring. The air is in an extremely turbulent condition and a continuous mixing of the air is taking place, some air from the rotor passing into the free atmosphere and new air being drawn into the vortex circulation.

It will immediately be apparent that the energy losses in such a circulation will be considerably higher than in a system with an established slipstream. Also, from the turbulent nature of the flow, it was to be expected that in flight, difficulties of control and serious vibration would be encountered. Anyone who has flown a helicopter in the vortex ring conditions will agree that these symptoms are much in evidence.

It is obvious that under such conditions the accurate measurements of the velocities in the vortex circulation are virtually impos-

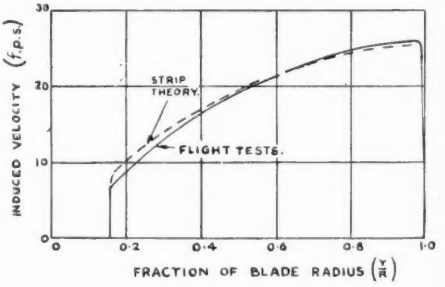


Fig. 2.
Comparison of experimental and theoretical axial induced velocity in hovering.

Fig. 3.
Typical vortex ring.
Photograph taken from rear fuselage of helicopter and giving view outboard of port undercarriage. Rate of descent about 500 ft./min. at zero forward speed.

sible. The fluctuations caused by the turbulence of the stream make it difficult even to identify the order of the velocity at any point in the circulation and, even if the measurement could be made, there is no relationship by which the conditions at the rotor disc could be obtained. However, some extremely valuable information on the nature of the vortex circulation has been obtained by the use of the smoke filament technique.

The nature of the vortex circulation is, of course, dependent on the flight conditions, *i.e.* power, rate of descent and so on. Thus, at the low rates of descent where high power and large pitch angles are used, the downward flow through the disc is high and the vortex ring may take in a circulation with a sectional diameter of the order of the rotor radius. Fig. 3 gives one of the first photographs of a vortex ring obtained in flight. This photograph was taken by a camera fitted on the rear fuselage of the Sikorsky R-4B. The pole carrying the smoke generator extends radially outwards from the top of the undercarriage oleo leg. It must be remembered in these photographs that the true position of the blade tip is about one ft. inboard of the end of the pole, but that the blade azimuth position relative to the camera may give the blade an apparent length considerably greater, or less, than the true rotor disc radius.

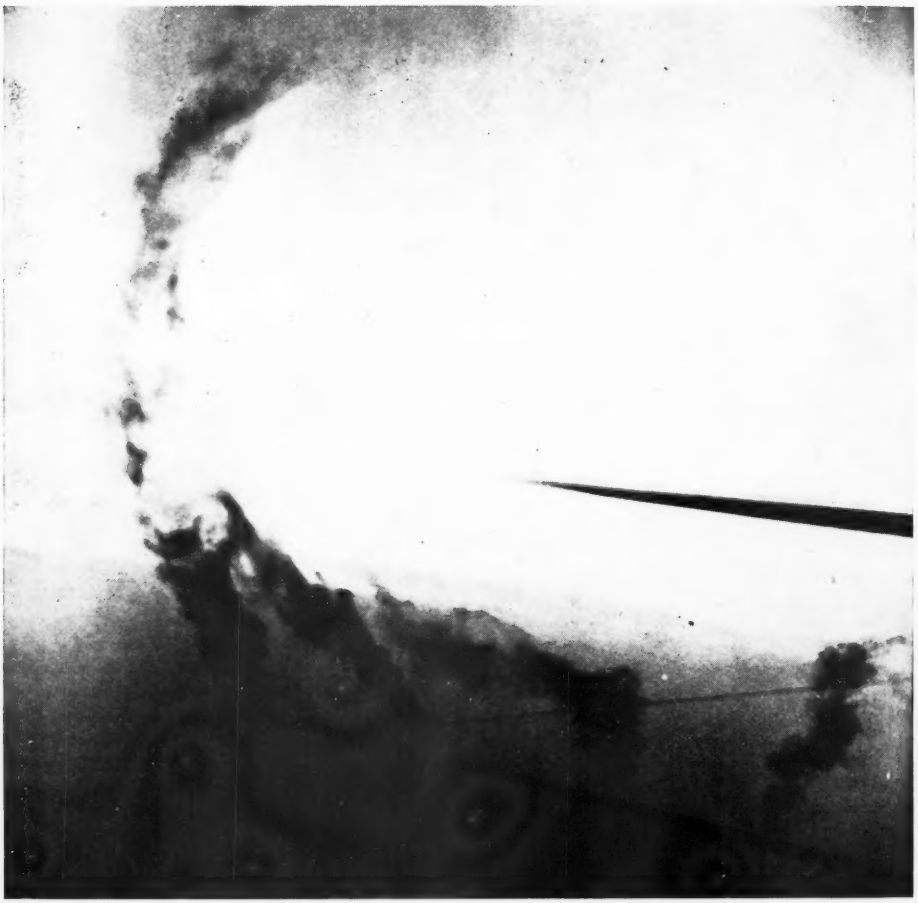


Fig. 4.
Vortex ring state.
Photograph showing technique of four simultaneous streamers to identify the vortex ring boundary. Rate of descent about 800 ft./min.

In Fig. 3 the smoke generator is situated at about 0.7 rotor radius. The photograph leaves no doubt that the air flowing through the rotor is largely maintained in the circulation. It also shows clearly that the vortex "ring" is really an ellipse, with the vertical direction forming the major axis, and this being considerably greater than the minor axis. Another important point to note is that the centre of the vortex ring in this condition lies below the rotor disc.

The encouraging results of the smoke visualisation of the vortex circulation with a single filament led to a more extensive application of this method. By sealing up the supporting pipe, except for four particular holes and by feeding the smoke from the generator into the end of the pipe, it was possible to obtain four simultaneous filaments and a much improved observation of the flow pattern could be made. In the following photographs (Figs. 4, 5 and 6) this method of using the four simultaneous streamers is used.

At a slightly lower value of pitch and power, with a correspondingly higher rate of

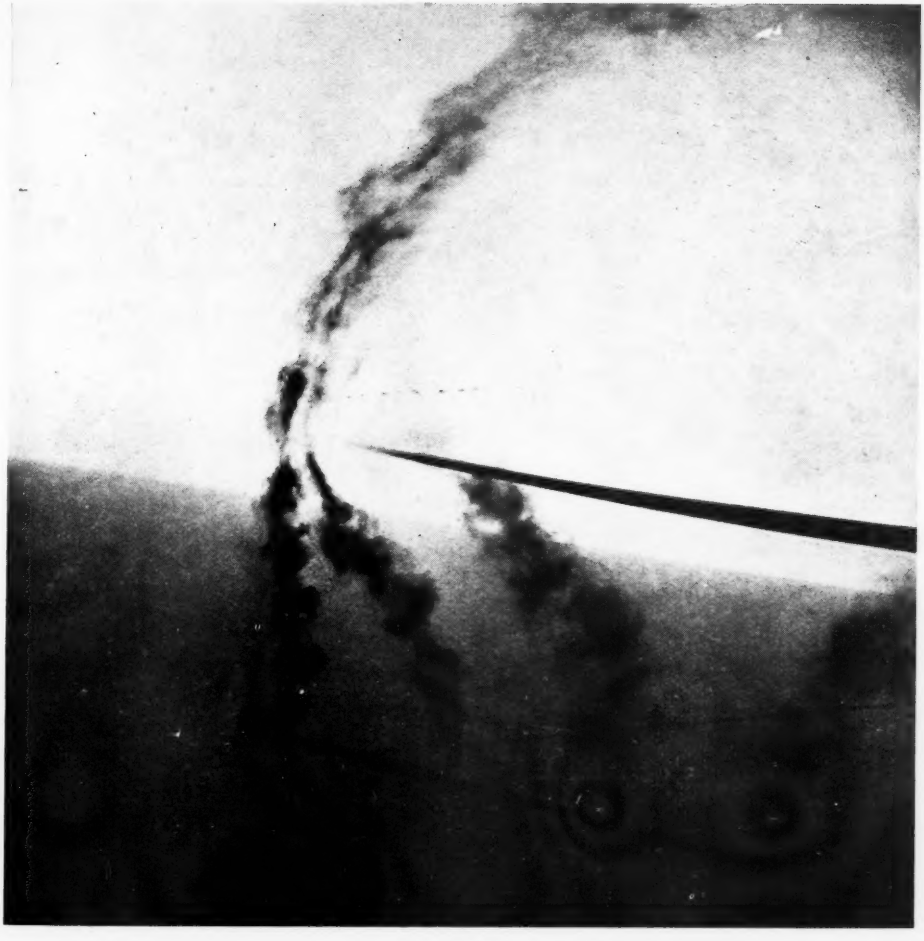


Fig. 5.
Vortex ring state.
Rate of descent about 1,500 ft./min. The vortex ring is now almost entirely above the rotor disc.

descent, than in the previous case of Fig. 3, we obtain the conditions of Fig. 4. The vortex ring is still very large but the influence of the greater rate of descent has positioned it higher relative to the rotor disc and the centre of the vortex now lies above the disc. It will be noticed from the inner streamer that the air below the centre of the rotor is not included in the circulation and that there may even be a tendency for the flow direction there to be upwards through the rotor. The convergence of the streamers in the upward flow outside the rotor radius is quite marked.

At a lower value of pitch and power, and correspondingly higher rate of descent, the size of the vortex ring diminishes and very little of it lies below the rotor disc. Fig. 5 shows such a case. Flow upward through the disc is beginning to be established at the outboard sections and upward flow is also taking place through the centre of the disc, although the axial velocities in this flow are very small. Very little of the air passing up through the outer section is drawn back

Fig. 6.
Auto-rotation boundary.
The helicopter is almost into auto-rotation. There is very little flow through the disc but a small circulation may exist above the rotor. Rate of descent 2,200 ft./min.

through the disc. This figure represents a state in the vortex ring conditions very close to the auto-rotation boundary.

The condition when the helicopter is just able to maintain auto-rotation at the same rotor speed corresponding to the various states discussed above, is shown in Fig. 6. This may be roughly taken as representing the ideal auto-rotation state. The rotor, in absorbing the energy from the air, is in a state of virtually no flow through the disc. There is probably still a vortex circulation above the rotor, but the velocities in such a circulation would be so small that for all general purposes the air above the rotor may be considered quasi-stagnant.

3.3 Windmill brake state.

In the last paragraph on the vortex ring conditions, the ideal auto-rotation state, which forms the boundary condition, was considered. Beyond this lies the windmill brake state of the ordinary propeller but on the helicopter the conditions do not form an

exact parallel. If the blade pitch is further reduced, the rate of descent increases and more work is absorbed from the air. For a helicopter there is no means of absorbing the power, as the rotor is free to over-run the engine and also, a constant thrust must be maintained to sustain the helicopter. The result is that the rotor speed builds up to some new value, where the energy is used in overcoming the profile drag of the blades at their higher rotational speed. There is, of course, to be taken into consideration the friction in the bearings and in the transmission to the tail rotor, as well as the profile drag of the tail rotor blades. Thus, as we progress further into the windmill brake state the rotational speed of the blades increases.

In the windmill brake state, the rotor is again working in a regular slipstream, the flow upward through the rotor having well defined streamlines. Momentum theory in the form of assumed constant induced velocity, or the application of strip theory, again becomes applicable.

In considering the boundaries between hovering and the vortex ring and between the vortex ring and windmill brake states, there is one important distinction to be noted. In hovering, there is a definite slipstream pattern of a laminar nature. In the case of the ideal auto-rotating rotor, the air is being brought to rest, there is still a vortex circulation in existence and a certain amount of turbulence or mixing will still be taking place. Thus, it would be expected in a practical case that the energy losses at the auto-rotation boundary would be greater than those at the hovering boundary.

3.4. Empirical estimation in vertical descent.

From the above discussion on the circulation round the aerofoil in the vortex ring state, it will be immediately obvious that a purely theoretical approach is extremely difficult, and, in fact, a satisfactory solution has not yet been attained. From the designer's point of view, some method of estimation must be used for the evaluation of performance and other considerations in these flow conditions.

In a lecture at the Anglo-American Conference, London, 1947, Hafner* suggested an empirical curve to cover the vortex ring conditions and presented this in a new form compared with the older methods of Glauert and Lock. The propeller and windmill brake states can be established theoretically and this, together with some earlier work on model propellers in the wind tunnel by Lock, and a few points from the Hafner gyroplane tests, formed the background from which his suggested curve was evolved. The curve is given by the ordinates of the form

$$\sqrt{(1/f_i)} = v_i \sqrt{(2\rho/p)}$$

and

$$\sqrt{(1/f_a)} = v_a \sqrt{(2\rho/p)}$$

where f_i, f_a are aerodynamic coefficients for the respective velocities,

v_i is induced velocity at the rotor disc,

v_a is axial velocity (rate of climb or descent for zero forward speed),

and p is the rotor disc loading.

Thus, for a given air density and rotor disc loading, the ordinates are directly proportional to the induced velocity and the rate of climb or descent. The presentation is a considerable improvement on earlier forms in that it gives much greater emphasis to the physical significance.

During the flight investigations of the flow conditions in the vertical descents, accurate measurements of the blade pitch, rotor speed, rate of descent, and so on, were taken. These have been analysed to give a full-scale comparison with the suggested curve. The most striking feature of these measurements was the time required in which to attain a steady rate of descent in the vortex ring. Again, the turbulence of the air made it difficult to maintain a true vertical descent and even when the pilot had attained considerable experience in this work, the development of small translational velocities rendered many of the runs useless. Even so, the measurements of rate of descent against pitch (and therefore power, the rotational speed being kept constant) under true vertical flight conditions gave considerable scatter. Further tests showed this to be mainly because of the time required for the attainment of a steady rate of descent as the vortex circulation built up. In the flight tests, the helicopter is brought into the vortex ring state at the appropriate pitch and power from a forward flight condition, by reducing the translational speed until zero is attained. An initial circulation of the air is set up around the rotor disc but two further developments then take place. The circulating air, as it repasses through the

* Rotor Systems and Control Problems in the Helicopter. R. Hafner. Aeronautical Conference, London 1947.

rotor, has its velocity further increased and at the same time the vortex circulation extends its influence radially, affecting more air. The net loss in energy causes the rate of descent to increase until the fully established vortex circulation is achieved. From this final condition, the relationships of blade pitch, power, rate of descent, and so on, were taken.

The flight results may be analysed in two independent ways. First, using the measured values of blade pitch angle, rate of descent, rotational speed of the blades and the actual blade shape, the thrust of the rotor can be evaluated in terms of these variables and the mean induced flow through the disc. For a steady rate of descent, the rotor thrust must equal the weight of the helicopter. Hence, by equating the thrust in terms of the induced velocity to the weight of the helicopter, the induced velocity can be obtained.

If v_i is the induced velocity at the rotor,
v_a the axial velocity of the rotor relative to the undisturbed air,
and θ_0 is the blade pitch,
the local incidence of the blade at radius r and with a rotational speed Ω is given by

$$\alpha = \theta_0 - (v_i + v_a)/\Omega r$$

The thrust for an element dr, where c is the local blade chord, is given by

$$dL = \alpha\,(dC_L/d\alpha)\tfrac{1}{2}\rho\,(\Omega r)^2 c\,dr$$

Putting the thrust for the disc equal to the weight of the helicopter

$$W = \int_{eR}^{R} \alpha\,(dC_L/d\alpha)\,\tfrac{1}{2}\rho\,(\Omega r)^2 c\,dr.$$

Second, the analysis may be made from power considerations. The engine power may be obtained from the maker's charts for the given manifold pressure, r.p.m. and so on as measured during flight. The total power input for the helicopter is given by the engine power and the potential energy due to the rate of descent. To obtain the induced power, the power absorbed by the profile drag of the rotor blades, the transmission and cooling loss and the power supplied to the tail rotor must be subtracted from the total power. The induced velocity can then be obtained directly from the induced power.

The ordinates $\sqrt{(1/f_i)}$ and $\sqrt{(1/f_a)}$ can then be obtained directly from the induced velocity and rate of descent for the given disc loading and atmospheric density. $\sqrt{(1/f_i)}$ is always taken as positive while $\sqrt{(1/f_a)}$ takes the same sign as v_a, *i.e.* negative for a rate of descent.

The results of the Sikorsky R-4B flight tests by the two methods of analysis are given in Fig. 7 for comparison with Hafner's presentation. The thrust and power analyses do not give absolute agreement. The thrust method is liable to be in error because of inaccuracy in the flight value of the blade pitch angle and to the effect of fuselage drag on the thrust. The latter term should be very small under the existing flow conditions. The power analysis inaccuracies are due to the possible variation in engine power for the actual engine compared with the type chart; assumption of the profile drag coefficient, although this has been fairly well established in previous flight tests; and estimation of trans-

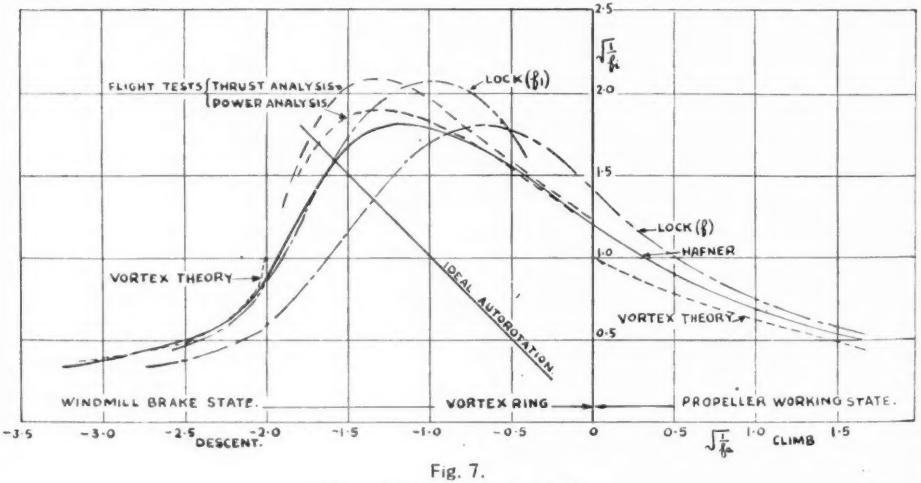


Fig. 7.
Induced flow in vertical flight

Fig. 8.
Forward flight.
Streamer passing through rear edge of disc giving a very interesting particle path trace through the vortices in the wake.

mission losses. Nevertheless, considering the experimental limitations, the two independent methods give curves of the same order and a mean value would probably be representative of the Sikorsky R-4B conditions.

The agreement with Hafner's suggested curve is very good towards the hovering end of the vortex ring state, but the induced velocities are about 10 per cent. greater towards the auto-rotation end. The Hafner curve based on autogyro work at this end is doubtful in this particular region, as the autogyro would have a well-developed slipstream. On the vortex ring boundary, turbulent circulation develops very rapidly, giving a very steep slope to the induced velocity curve. The Hafner curve has about the correct gradient but the flight tests suggest that the turbulent vortex circulation covers a greater range than had been anticipated. At the same time, it is difficult to see why the Hafner curve should have been drawn below the negative arm of the simple vortex theory.

3.5. Forward flight.

The treatment of induced velocity in forward flight is difficult, because of the dependence on the azimuth position of the blade. The tilt of the rotor in forward flight and the flapping angle of the blades gives an axial component of the air flow varying with azimuth position of the blade.

Preliminary flight tests have established a satisfactory technique, using smoke streamers, which will give the general nature of the flow and the order of the variations in induced velocities. This work is only in its initial stages but indications have already been obtained of upflow in front of the rotor and that there is an increase in downward flow towards the rear of the disc.

Figure 8 gives a picture of a smoke streamer passing just above the rotor and down through the rear edge. This photograph was obtained from another helicopter flying in formation. The smoke generator is on the end of a long wire trailed from a slow flying Fieseler Storch aircraft. A most interesting trailing vortex system can be seen in the wake. It may be thought at first sight that this is a tail rotor effect, but correlation of the distances between the vortices with the forward speed of the helicopter and rotational speed of the blades shows it to be the tip vortices shed from the main rotor blades. These tip vortices follow the surface of an oblique cylinder while the smoke traces the particle path through them.

To obtain a knowledge of the variation of the induced velocity across the longitudinal direction of the disc, a number of simultaneous streamers will be necessary, as only with a complete picture of the flow through the disc under a particular condition could any degree of accuracy be assured.

The information from such a series of tests will not only be of value from the point of view of induced velocity of the single rotor helicopter, but would give the nature of the flow in the neighbourhood of a tailplane installation. It would also be of immense

value for the design of tandem rotor helicopters.

4. PERFORMANCE TESTING.

The performance testing methods developed for the helicopter do not vary much from those adopted on the orthodox fixed-wing aircraft. One of the main difficulties is the accurate measurement of the low air speeds possible in helicopter flight. Below 20 m.p.h. the differential pressure of an ordinary pitot-static system has become so small, that it is difficult to record with any degree of accuracy. In addition, the flight conditions are comparable with the magnitude of the rotor flow and interference effects become large.

As the speed range of the helicopter is small, it is quite convenient to plot a speed polar diagram from which the top speed, best climbing speed and corresponding rate of climb, and vertical flight performance can all be obtained accurately.

4.1. Position error.

It is difficult on a helicopter to find a position for the pitot-static installation which will not be affected by the flow from the rotor. For the Sikorsky R-4B, with the pitot situated between the top of the fuselage and the rotor and the static vent on the rear fuselage, the position error varies in a complex manner with change in air speed, power and centre of gravity position. This installation is entirely unsatisfactory for performance work and all the measurements on this type of helicopter had to be made by using trailing instruments.

In general, the most satisfactory position for a pitot-static system on a helicopter would appear to be projecting from the nose and as low as possible to give the greatest speed range free from rotor interference.

4.2. Effect of vertical currents.

A trailing pitot-static (80 ft. of line), universally mounted on the end of the cable to allow it complete freedom to take up the direction of the relative air flow, proved very satisfactory in use. This eliminated any position error.

Rates of climb and descent measured by the pitot-static are subject to errors introduced by random vertical currents in the atmosphere. For a slow-flying aircraft the climb takes place in a correspondingly smaller section of the atmosphere, thus on the helicopter the effect becomes quite marked. Of course, the poorer the performance the more important is this effect when evaluating the measured performance.

The effect of these random vertical currents on the performance of the Sikorsky R-4B was demonstrated by taking height—time histories for a series of partial climbs made within a few minutes of each other. Particular attention was given to constant engine and temperature conditions, and the flight made in what would be considered ideal weather for performance testing. In fact, all the conditions were identical, except for the variation in the helicopter weight due to fuel consumption, amounting to less than one per cent. The measured rates of climb varied by as much as 100 ft./min.

While the aneroid method may be sufficiently accurate for routine tests, it is not satisfactory for research performance work. The effect of the vertical currents can be eliminated by using a flight path recorder. This instrument was used on slow-flying fixed-wing aircraft many years ago and did not need any modification for use on the helicopter. It measures the true air speed by an airlog and the angle of climb (relative to the air) by a pendulum. The results are obtained on continuous trace film records.

Comparing the accuracy of the two methods of performance measurement on the Sikorsky R-4B, the root mean square error for the aneroid method was 90 ft./min. while for the flight path recorder it was only 40 ft./min. on $\frac{1}{2}$ min. runs and 25 ft./min. for runs of 2 minutes. It is interesting that the magnitude of the vertical currents, represented by the errors in the aneroid method from the mean curve, agrees with the measurements made by Alston many years ago.

4.3. Vertical flight.

The measurement of air speed above 15 to 20 m.p.h. can be made quite satisfactorily by pitot-static or flight path recorder, but below this value there is no simple accurate method available. Vertical flight can be indicated by a trailing line with a weight at its lower end. The slightest development of speed in any direction is immediately apparent as a bowing of the cable.

Steady vertical flight in the range from hovering to about 2,000 ft./min. rate of descent is difficult because of the instability of the flow in the vortex ring state. Thus, for the Sikorsky R-4B where there is insufficient power to hover at normal load, it is extremely

difficult to measure the full power zero air speed performance.

4.4. PERFORMANCE TECHNIQUE.

There is no need to elaborate further on the measurement of level speeds and rates of climb, or descent, as the technique is similar to the performance testing methods adopted on the orthodox aircraft.

In take-off and landing considerations several problems are introduced if the helicopter has not sufficient power to enable it to hover free from ground cushion assistance. In such a case, the helicopter is flown vertically upward within the ground cushion to the limiting condition of this effect and then translation speed must be gained before a climb away can be made. Thus, from the point of view of clearing obstacles, there is an operable take-off distance limitation. This can be considered as similar to the take-off measurement on an ordinary aircraft, where for the helicopter there is no ground run but the "transition" distance still exists. Similar considerations apply to the landing of the under-powered helicopter.

Such conditions should only be a transitory phase in helicopter development and all future helicopters should be able to take-off and climb away or to descend and land from the recognised 100 ft. limit with zero ground distance. Nevertheless, one important feature will still require attention, namely, the performance, particularly the landing, of the helicopter following power failure. The success of such a landing in auto-rotation—if a restricted space is considered—depends on the pilot's utilisation of the inertia of the blades to give a temporary increase in thrust. The designer has control over these conditions by arranging for sufficient weight in the blades to enable the helicopter to achieve a hovering condition before touch-down. A recent paper by Fitzwilliams[12] gives an excellent treatment of this problem.

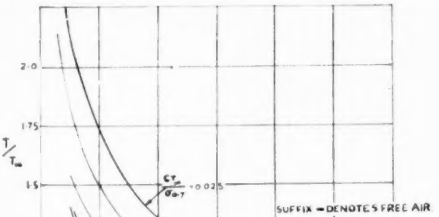


Fig. 9

Ground effect on thrust at constant power.

The effect of engine failure during take-off or landing must also be considered. On present helicopters there is a small height range (greatest in vertical flight) in which, should engine failure occur, there is not sufficient height to develop complete auto-rotation on the one hand and, the helicopter is too high to permit immediate utilisation of the kinetic energy of the blades, on the other. Increased inertia of the blades will enable this danger region to be reduced at both ends of the range, but if it cannot be entirely eliminated some modification to take-off and landing technique may be required, even though there is sufficient power for vertical climb.

4.5 GROUND EFFECT.

As discussed in the previous paragraph, it is anticipated that all future helicopters will be capable of hovering free from the ground cushion. In cases of operational over-loading this may not be true and an appreciation of the ground cushion effect is necessary.

In hovering flight the helicopter derives its lift by imparting a downward velocity to the air passing through the rotor. When the rotor is brought into proximity with the ground, the slipstream is not allowed to develop in its free-air form. The reflection from the ground causes a redistribution of the flow and affects the pressure and velocity of the air at the rotor. This affects both the thrust and power absorption of the rotor. The main thing the designer wants to know is the variation in lift obtained at constant power.

A satisfactory theoretical treatment of this condition has not yet been established, but a semi-empirical method has given us the information we require. The main parameters are the relative height above the ground (ratio of height to rotor diameter) and the mean lift coefficient for the blade, used in the form of a blade loading coefficient $C_T/\sigma 0.7$. Other parameters such as rotational speed, number of blades, profile drag and actual shape of blade have only a minor influence.

Applying this method to an analysis of the many available model tests of the ground effect, a family of curves was obtained in terms of the main parameters giving the

increase in thrust for constant power conditions. A series of flight tests made on the Sikorsky R-4B confirmed the application of these results to the full-scale helicopter.

It will be seen from the curves (Fig. 9) that the ground effect is negligible at a height of one rotor diameter above the ground. The thrust at constant power increases rapidly close to the ground. For helicopters of the present day with a rotor clearance of about a quarter of the rotor diameter and a blade loading coefficient of 0.06, the ground cushion gives initially a 35 per cent. increase in thrust, 20 per cent. increase at a height of 20 ft. and 10 per cent. increase at 30 ft. This typical example shows the magnitude of the ground cushion assistance.

5. STABILITY AND CONTROL.

5.1. GENERAL CONSIDERATIONS.

Before discussing the handling qualities of the helicopter in their particular aspects, it is well to consider briefly the general characteristics of the control of a flying machine. The general movement of any body in space is determined by its six degrees of freedom, namely, the linear movements along, and the angular rotations about, three mutually perpendicular axes. While it is true to say that for complete control in space the body must be provided with six independent controls. satisfactory control can be achieved with fewer.

If we consider the development of the orthodox fixed-wing aircraft, we find that control is almost universally obtained from the three angular movements, produced by the elevator, ailerons and rudder respectively, and linear movement along the longitudinal axis of the aircraft by propeller or jet thrust. By suitable combination of the effect of these controls—and of their secondary effects—satisfactory behaviour of the aircraft is obtained and complicated aerobatic manœuvres can be performed.

The development of the helicopter with its various rotor configurations has evolved various methods of control. In general terms, the controls provide linear movement along the normal axis (thrust of the rotor(s)); angular movement about the normal axis; combined linear movement along the longitudinal axis with rotation about the lateral axis; and combined linear movement along the lateral axis with rotation about the longitudinal axis. This system varies in certain important aspects from the controls of the orthodox aircraft, but in their presentation to the pilot, it is possible to achieve a fairly close analogy between the two systems.

For the present purposes, we shall consider only the single rotor helicopter. For those not accustomed to the controls of the helicopter, a brief description of the method of control is given.

5.2. SINGLE-ROTOR HELICOPTER CONTROLS.

5.2.1. *Collective pitch.*

The lift produced by a rotating blade depends on the local incidence and the square of the velocity at each blade element. The local incidence in turn depends on the geometric pitch setting of the blade and the resultant flow through the disc. By increasing the pitch of the blades, at constant rotational speed, the thrust of the helicopter rotor is increased. The pilot, by means of the collective pitch lever—usually a long lever located on the pilot's left—has a direct control over the resultant thrust of the rotor. As the pitch is increased, the drag of the rotor blades also increases and the rotational speed is maintained by increase in engine power. There is an inter-connection between the collective pitch and the throttle; increasing the pitch automatically increases the throttle opening. A twist-grip throttle, situated at the end of the pitch lever, acts as an adjustment to the throttle setting given by the pitch lever inter-connection and allows limited independent control of the rotational speed of the blades.

5.2.2. *Control stick.*

The control stick, by positioning a universally-mounted swash plate (or similar system), causes the pitch of each blade to follow a sinusoidal variation at rotor frequency. This pitch variation is termed "cyclical pitch" and the azimuth phase position and amplitude are determined by the stick position.

When cyclical pitch is applied to the blades, the lift tends to be greater where the pitch is higher. For freely flapping blades (hinged about a horizontal axis) the motion of the blades depends on the moments of the lift, centrifugal force and blade weight about the flapping hinge. It can be shown that the maximum blade position occurs 90 degrees later than the maximum cyclical pitch. The flapping motion results in a tilting of the rotor disc plane about the main shaft and as the resultant lift vector of the rotor is approxi-

mately coincident with the effective rotor disc axis, control in the plane of the resultant lift vector is achieved.

It should be noted that the tilting of the rotor disc produces only a linear acceleration in the plane of the resultant lift vector. As the blades are hinged at the rotor centre, moments cannot be transmitted to the fuselage. If the C.G. of the helicopter coincided with the rotor centre no angular movement would be produced, but as the C.G. is a considerable distance below the rotor, tilting of the disc displaces the thrust vector giving a moment about the C.G. of the helicopter.

5.2.3. *Tail rotor.*

The methods of obtaining directional control of the helicopter and of countering the torque reaction, form the major influence in the single rotor helicopter layout. Here, only the well-known Sikorsky type (vertical propeller at the tail) is considered. This tail rotor acts as a means of directional control in addition to its function of torque reaction correction. Operation is by means of orthodox "rudder" pedals controlling the pitch setting of the tail rotor blades. The pilot's pedals are set central with the correct pitch applied to counter the torque reaction at cruising conditions.

It should be noted that this control produces a lateral thrust and a moment about the helicopter C.G. If a pure couple about the normal axis is desired, an equal and opposite thrust must be produced by a lateral tilting of the main rotor.

5.3. GENERAL HANDLING TESTS.

5.3.1. *General impression.*

The general handling qualities of an aircraft are determined mainly by the pilot's impressions. With the object of putting these handling qualities on a scientific basis, the pilot is given a series of manœuvres to perform from which the general nature of the stability and control characteristics of the helicopter can be assessed. It is important that for each flight the loading, centre of gravity position, and so on, should be recorded, as the flying characteristics may vary considerably with these parameters.

It is interesting to record that, to pilots experienced on orthodox fixed-wing aircraft, the helicopter is an entirely new type of flying machine and as such, it requires time for them to become accustomed to the necessary co-ordination of the controls and to the different conditions of flight made possible by the unique characteristics of the helicopter.

5.3.2. *Take-off and landing.*

The take-off and landing of the helicopter are inherently a much easier process than on the fixed-wing aircraft. For take-off, the pitch and power are increased gently and ample time can be taken to obtain the correct positioning of the controls to prevent any translational motion or turning on leaving the ground. Care must be taken to ensure that there is sufficient power at each stage of the pitch increase to prevent any decrease in the rotational speed of the blades, which might result in the helicopter sinking back on to the ground. When the required height above the ground has been attained—on the Sikorsky R-4B lack of power limits vertical ascent to a height of about 16 ft. (in the ground cushion) at normal load in zero wind —the stick is eased forward to gain translational speed.

Increase in wind strength improves the take-off immensely, because of the reduction in the power required to maintain height at zero ground speed.

Take-off in cross-wind conditions, although not generally necessary, presents no difficulties. The power is increased more rapidly during the latter part of the throttle operation to ensure a vertical take-off and prevent any risk of the wheels making contact with the ground should the helicopter drift sideways. In strong cross-wind conditions, considerable tail rotor displacement is required to prevent drift.

The normal method of approach is similar to that in an orthodox aircraft. A circuit of the landing area is made with a gradual descent into wind to approach the selected touch-down point. The circuit may be made at any height above the "safety height" for auto-rotation procedure. Height is then reduced at constant air speed, corresponding to the speed for minimum power, a close check being kept on the air speed indicator since the attitude of the fuselage gives no guide as to the air speed.

At a height of about 30 ft., the air speed and rate of descent are gradually reduced simultaneously and the power and pitch increased, until the helicopter achieves a hovering condition just above the ground. This hovering condition is maintained until the ground effect is fully established. The pitch is then gently reduced until the wheels

make contact with the ground, when rapid pitch reduction allows the helicopter to settle.

In the event of engine failure, a safe landing can be made in auto-rotation, although a certain amount of judgment on the part of the pilot is necessary. The approach is made at the speed for minimum rate of descent. At a height of about 30 ft. a hold-off is made by pulling back on the control stick, reducing the air speed towards zero and arresting the rate of descent. Close to the ground the pitch of the main rotor blades is increased quickly, the inertia of the blades being utilised to maintain a high rotational speed at a high lift condition for sufficient time to reduce the touch-down impact considerably. If a smooth stretch of ground is available, a landing with forward speed—similar to that on an aeroplane—can be made without alteration of the pitch setting.

5.3.3. *Longitudinal control tests.*

The longitudinal control of the Sikorsky R-4B is considered quite light and effective. The main criticisms are the lack of proper trimming devices and the continual vibration transmitted to the stick in the form of a stirring motion. The forces to move the stick are very light and consist mainly of the inertia forces of the control system—the faster the control is moved the higher the forces. When the stick has reached its new position the force virtually disappears. While the control of the helicopter is good, the stability is poor and the pilot has to make continuous corrections, with the result that a fair amount of concentration is necessary, leading to fatigue if flight is continued for any length of time. From the point of view of giving an indication of the stability characteristics, the pilot may be asked to make some particular measurements such as those described in the following tests.

These tests are started from straight and level flight. The pilot displaces the stick from its trimmed position and releases it. The resultant behaviour of the helicopter gives an indication of its stick *free* stability. For the Sikorsky R-4B, if the stick is displaced forward and released, the helicopter dives and the stick moves farther forward as the dive steepens. If the stick is displaced backward and released, the helicopter develops a climb in an increasing nose-up attitude, from which recovery action must be taken to prevent the development of a tail slide. Thus, the stick *free* stability is aperiodically divergent.

In a second series of tests, the pilot displaces the stick and returns it to its initial trimmed position. The behaviour of the helicopter in this case indicates the stick *fixed* stability. Under these conditions, the Sikorsky R-4B develops an oscillation, the period and damping factor depending on the initial conditions of air speed, power, and so on. The results of an investigation of the stick-fixed stability with complete instrumentation are dealt with in detail in a later section.

5.3.4. *Lateral control tests.*

The series of lateral control tests is of a similar nature to the longitudinal control tests. The lateral control is also light and effective but has been criticised by pilots as too sensitive. It is interesting to note that equal lateral or longitudinal stick displacements apply cyclical pitch of approximately equal amplitudes. Thus, the tilting of the disc laterally or longitudinally is similar for a given sideways or fore and aft stick movement. The rolling moment of inertia is only a tenth of the pitching moment of inertia, hence the angular movements of the fuselage occur more rapidly in roll and the tendency to over-control the helicopter laterally is accentuated.

Lack of any form of lateral trim makes the stick-free lateral characteristics difficult to evaluate. On the Sikorsky R-4B, if the control stick is displaced laterally and released the initial roll is damped out, but the control is out of balance to the left and a turn to port begins and this soon develops into a steep spiral dive.

With the control displaced laterally and fixed in its initial position, a combined lateral and directional oscillation ensues. The period is of the order of 2 to 3 sec. and the motion is rapidly damped in forward flight.

5.3.5. *Loss of control.*

The most undesirable characteristic experienced in the flight tests of the Sikorsky R-4B was a loss of control during vertical descent. This occurs frequently in zero forward air speed descents at values of collective blade pitch greater than 7 degrees and is always obtained under these conditions, if the pitch is reduced. The loss of control takes the form of a nose-down pitching of the helicopter and full backward movement of the control stick cannot prevent the speed increasing rapidly to about 40 m.p.h., with consequent loss of height in regaining control.

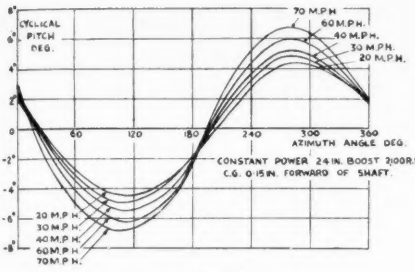


Fig. 10.
Cyclical pitch distribution in forward flight v. air speed.

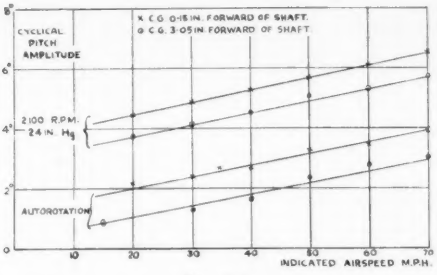

× C.G. 0·15 IN. FORWARD OF SHAFT.
○ C.G. 3·05 IN. FORWARD OF SHAFT.
CYCLICAL PITCH AMPLITUDE
2100 R.P.M. 24 IN. Hg
AUTOROTATION
INDICATED AIRSPEED M.P.H.

Fig. 11.
Cyclical pitch amplitude v. air speed.

The possibility of transient stalling of the blades over part of the rotational arc was considered, but this was disproved by a series of tests in which the loss of control could be suppressed by increasing pitch and power immediately the pitching motion began to take place.

The explanation lies in the general unstable nature of the flow in the vortex ring state. The rate of descent curve against translational speed takes the form of a cusp at zero speed and small changes in translation give considerable redistribution of the flow under the rotor. These vertical changes in flow have a considerable effect on the fuselage pitching moment. Flight tests with smoke streamers and with wool tufts have shown that reversals of flow over the rear fuselage take place as the pitching motion is set up. The development of backward velocities appears to have a particularly serious influence on the nose-down pitching of the helicopter.

This loss of control due to the fuselage pitching moments in the unstable flow condition should be borne in mind, especially when considering the fitting of a tailplane to the helicopter.

5.4. Static Longitudinal Stability.

The static longitudinal stability of any flying machine is one of the main features which influences the pilot's opinions of its flying qualities. In simple terms it may be explained as the changes required in the stick position and in the stick force required to change from one trimmed speed to another. For example, to increase the speed the pilot moves the stick forward and if the trimmed position of the stick at the higher speed is forward of its initial position, the aircraft is said to be stick-fixed statically stable. Also, if at the higher speed the force in the stick tends to move the stick back, the aircraft is stick-free statically stable. If the machine is unstable stick-fixed, the pilot, after an initial forward movement of the stick, would have to return it to a position aft of its initial position to prevent the speed exceeding the desired change. Similarly, for stick-free instability the force in the stick at the higher speed would tend to move the stick forward.

Of the two static stabilities, the stick-*free* conditions have the greater influence on the pilot's impressions of the aircraft. The Sikorsky R-4B helicopter may be considered about neutrally stable stick-free. The forces in the stick are very light—once the inertia forces are damped out—and are completely obscured by the vibration, so that accurate measurement is difficult.

Stick-fixed stability conditions lend themselves much more readily to flight measurement and are treated here in terms of the cyclical pitch applied to the blades for the various conditions. If we consider changes in speed of the helicopter under constant power conditions, the change in relative velocities over the advancing and retreating blade necessitates a change in the cyclical pitch distribution (or feathering of the blade) to equalise the lift. This is offset, to some extent, by the equivalent feathering produced by the change in tilt of the disc required to overcome the drag at the new condition.

Figure 10 shows the cyclical pitch applied to the Sikorsky R-4B at various forward speeds for a given constant power condition. It will be seen that the phase azimuth position is almost constant. The cyclical pitch amplitude is plotted against speed for power-on conditions and auto-rotation, and for two centre of gravity positions in Fig. 11. These curves are analogous to elevator trim curves

for the fixed-wing aircraft. The case of the helicopter is not as simple as the orthodox aircraft as change of power alters the torque reaction and therefore, the thrust of the tail rotor and for equilibrium a change is required in the lateral tilt of the rotor, equivalent to a change of the phase location of the cyclical pitch curves.

The stick displacements to trim the helicopter are small and the entire speed range can be covered with a movement of just over one inch, while the available movement is about 12 inches. Thus, the R-4B may be said to be only slightly statically stable.

Turning to the curves for the Sikorsky R-4B, we find that these take the form of approximately straight lines parallel to each other for the various conditions. Here we find one of the major differences from the static stability of orthodox aircraft, namely, no change in stability with change of centre of gravity position. Thus, such well-known functions as the "manœuvre point" and "neutral point" of the orthodox aircraft have no equivalent for the single rotor helicopter. In fact, for the latter, changes of centre of gravity position have no effect on the flying characteristics of the helicopter and appear only as changes in the trimmed position of the controls. It appears that this "simplification" of the stability will only be found on the single rotor helicopter and that on tandem or multi-rotor configurations dependence on the centre of gravity position will again become critical.

5.5. Dynamic stability.

The recent flight measurements of the longitudinal dynamic stability of the Sikorsky R-4B constitute one of the most interesting items of research in the present phase of helicopter development. It is particularly important in view of the complete lack of accurate measurements in this section of helicopter work, and a fairly full account of this work is included.

5.5.1. *Instrumentation.*

For the dynamic stability tests, an automatic observer was essential. Desynn transmitters were fitted to the two rods operating the swash plate. Each of these transmitters operated two receivers, one in the automatic observer and the other in front of the pilot. Desynn transmission was also used for the collective pitch of the main rotor blades and of the tail rotor blades.

As the aircraft air speed system does not give accurate speed measurements, a special trailing pitot-static system was used to obtain pitot and static pressures free from interference from the helicopter. The pitot-static was mounted in a gimbol and ball race, allowing complete freedom to swivel in any direction. A differential pressure gauge (used as a more accurate type of low reading air speed indicator), an altimeter and a rate of climb meter were connected to this trailing system. These instruments were used primarily to give the initial conditions and it will be appreciated that in the phugoid motion, the indications may lag behind the actual helicopter conditions.

An electrically-driven free gyro was installed to give change in attitude. The gyro could be reset from the observer's position at the beginning of each run.

Normal acceleration, engine manifold pressure, engine and rotor r.p.m. and a timing clock were also included in the automatic observer.

5.5.2. *Flight technique.*

The essential feature of the flight testing technique was to make certain that the control was returned to, and maintained in, its initial position, after displacement. It was not considered advisable or practicable to fit a rigid clamp to the controls because of the vibration. During the tests, the pilot obtained the steady initial conditions required and noted the desynn reading of the stick position. The control was displaced smartly, returned to its initial position and firmly held by the pilot. The firmness of the pilot's grip was the main feature in maintaining the trimmed position and the use of the desynn indicators to ensure that the stick did not wander from its initial position was a secondary consideration. Attempts were made to maintain constant control position by holding the stick loosely and trying to keep the desynn readings constant, but this led to continuous over-correction.

Simultaneous time histories of the flight conditions and control positions were obtained during the longitudinal phugoid motion. On analysis of the films, only measurements in which the control position variations were very small were used.

The power-on tests were made from straight and level flight at constant height, except at very low speed where there is insufficient power available to maintain height.

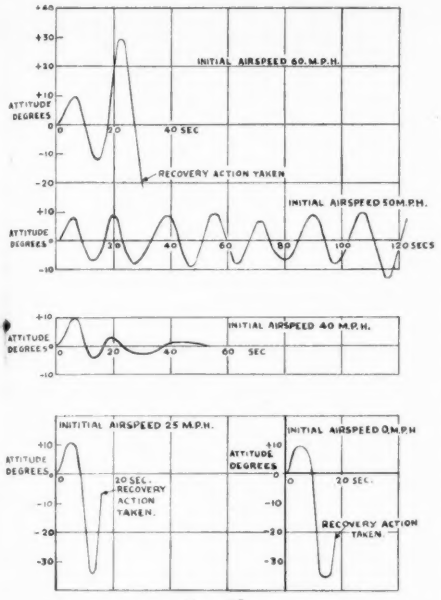


Fig. 12.
Typical attitude—time histories. Sikorsky R-4B. Power-on conditions.

Tests in auto-rotation were made mainly at $2\frac{1}{2}$ degrees pitch, but a few measurements were made at 4 degrees pitch (the rotor speed being correspondingly lower).

Measurements were made for three centre of gravity positions covering a range of six inches.

In each condition measurements of the phugoid motion were made for forward and for backward initial displacement of the stick.

5.5.3. *Method of analysis.*

Air speed cannot be measured with sufficient accuracy, particularly at the lower end of the speed range, to define the longitudinal motion. The variations of the helicopter attitude obtained from the attitude gyro records were used to give the period and damping factor.

The envelope of the attitude-time oscillations was assumed to be of the form

$$\theta = \theta_0 e^{\lambda t}$$

where θ represents the attitude angle and λ the damping factor.

In some conditions the phugoid motion was so rapidly divergent that it was not possible to draw the envelope curve. The value of λ in such cases was obtained by using any two peaks of the phugoid θ_1 and θ_2 occurring at times t_1 and t_2 according to the formula

$$\lambda = \{ \log_e (\theta_2/\theta_1) \} / \{ t_2 - t_1 \}$$

The possible length of run during the phugoid motion varied considerably with the initial conditions. In the power-on conditions at about 60 m.p.h. the divergence necessitated recovery action during the third period of the oscillation. At 50 m.p.h. the motion had a constant amplitude and it could be continued for any desired length of time. Generally five or six consecutive periods were measured, but in several cases the motion was continued for ten periods. Between 20 and 30 m.p.h. the rate of divergence of the oscillation was so great that, for the smallest practical displacement, the speed became negative during the second period. As the "tail slide" developed from this condition a serious nose-down pitching occurred. The angular velocity during this was of the order of 20 deg. per sec., but cases of over 40 deg. per sec. have been measured. Even with recovery action taken on the controls, the helicopter had achieved nose-down attitudes of as much as 60 deg. when the rotation stopped.

Below 20 m.p.h. it was impossible to obtain satisfactory phugoids from a backward displacement of the stick, as the nose-up attitude of the helicopter reduced the air speed below zero and the nose-down pitching occurred during the first period. For a forward displacement, one complete period could be obtained before the pitching occurred and this allowed an evaluation of the damping factor.

In auto-rotation, the dynamic stability does not vary largely with speed. The main limitation to the length of run is the height available for obtaining steady conditions, carrying out the test and regaining level flight. At very low air speeds, a simple phugoid was difficult to obtain as the longitudinal motion soon developed into a rotary swinging motion like that of a descending parachute.

5.5.4. *Results.*

Typical attitude-time records during the power-on dynamic stability tests are given in Fig. 12. These have been selected for approximately the same initial displacement and illustrate the development of the phugoid motion under various conditions.

The dynamic stability characteristics are given in terms of the damping factor and periodic time of oscillation in Figs. 13a and b

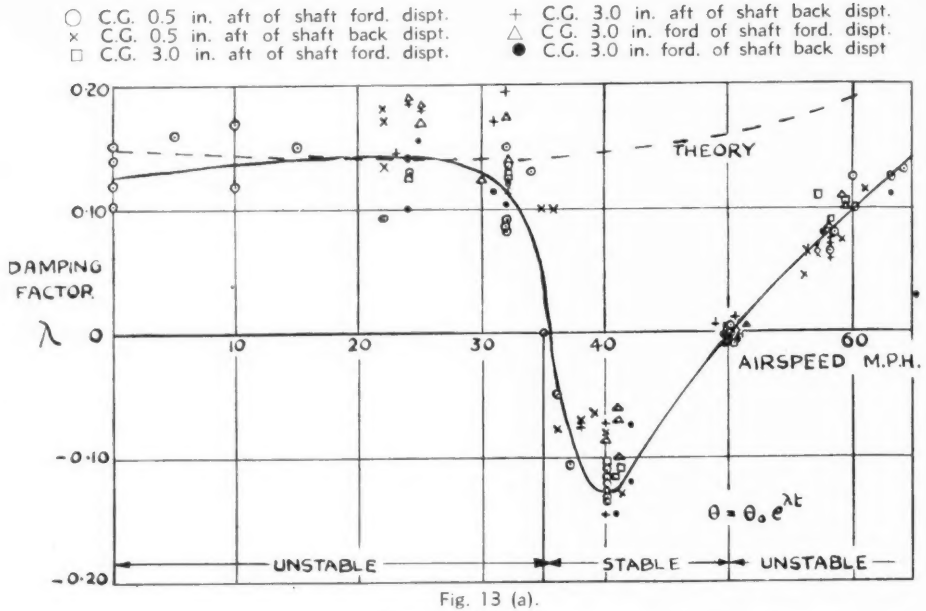
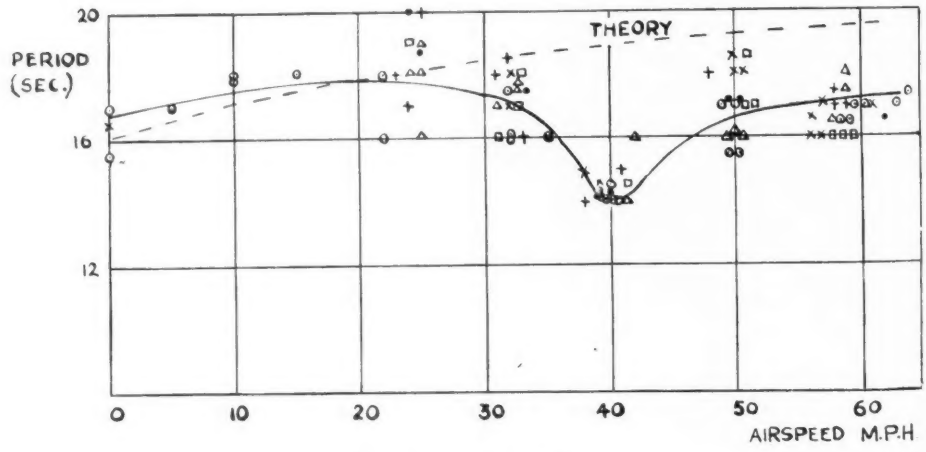


Fig. 13 (a).
Damping factor λ.



Periodic time of oscillation.

Fig. 13 (b).
Dynamic stability—power-on conditions. Sikorsky R-4B.

respectively for the power-on conditions. It will be seen that there are large changes in the stability with speed. The most surprising effect is the sudden change at about 35 m.p.h. from a rapidly divergent oscillation to an equally rapidly damped one within a speed range of a few m.p.h.

The auto-rotation results are given in terms of the damping factor and period of oscillation in Figs. 14a and b. In auto-rotation there is little change in the dynamic stability with speed, an improvement taking place as the speed increases.

The experimental points for the various centre of gravity positions are shown in the figures. No difference in the behaviour of the helicopter could be detected with change in centre of gravity position. There is no difference in the motion set up from an initial forward displacement of the stick compared with that from initial backward displacement.

5.5.5. *Comparison with theory.*

The theoretical estimation of the dynamic stability characteristics, for comparison with the flight measurements, has been based on the work of G. J. Sissingh. The lack of data on the rotor derivatives makes such a theoretical treatment difficult to evaluate. An outline of the theory is given in Appendix I. The estimated stability characteristics are included in the figures of the flight measurements.

The comparison in the power-on conditions is very good at low air speed but the theoretical estimation does not show the rapid improvement indicated by the measured results. This discrepancy is thought to be due

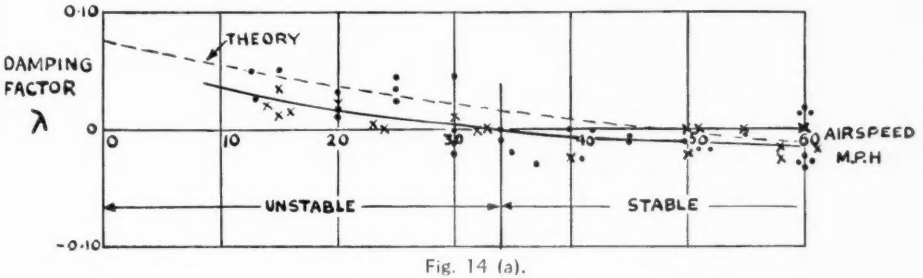
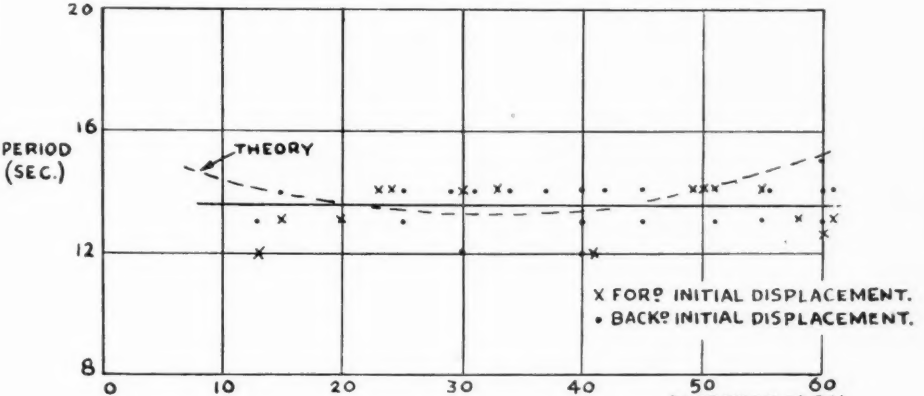


Fig. 14 (a).
Damping factor λ.



Periodic time of oscillation.

Fig. 14 (b).
Dynamic stability in auto-rotation. Sikorsky R-4B.

to the influence of the rotor slipstream on the fuselage pitching moments. In hovering, or at very low air speed, the slipstream is roughly uniform and acts downwards on areas of the same order fore and aft of the lateral axis. As the air speed increases, the slipstream is inclined backward and affects only the fuselage aft of the centre of gravity. As the air speed is further increased, the slipstream affects less and less of the fuselage surfaces and also, velocity of the air in the slipstream approaches that of the free air. Thus, large changes in fuselage pitching moment with forward speed would be expected, in addition to changes with attitude. The effect of the induced flow from the rotor on the fuselage pitching moment during the motion could not be allowed for in the theoretical treatment.

In the auto-rotation tests, the comparison of the estimated and measured quantities is very good throughout the speed range. A better agreement in auto-rotational flight than in the power-on conditions was to be expected, because of the lack of induced flow effects and because there is more available wind tunnel data on rotor derivatives for low values of blade pitch.

5.5.6. *Effect of gusts.*

It is particularly noticeable in flying the helicopter that atmospheric gusts have very much less effect than that usually experienced on fixed-wing aircraft. This is a feature of the flapping blade system, which may be regarded to some extent as an "automatic gust alleviator." In the fixed-wing aircraft, the response of the aircraft in reducing the gust loads, etc., operates on the inertia of the entire aircraft, while in this case only the inertia of the blades is concerned.

Several flights were made in rough air conditions at power and speed conditions corresponding to serious instability. As in the phugoid tests, it was difficult to avoid small stick movements during the run but these could be kept to about the same order as in the phugoid measurements. The helicopter could be flown through quite severe gusts without showing any tendency to develop the unstable oscillations.

A theoretical investigation of the effect of gusts on the flapping bladed rotor shows this effect to be of an entirely different nature to that produced from longitudinal stick displacement. Whereas a fore-and-aft movement of the stick produces a longitudinal tilting of the disc, the main effect of flying straight into a gust gives a direct effect on the lift and a lateral tilting of the disc. This is, of course, due to the phase difference between the change in incidence of a hinged blade and the position of maximum blade displacement. There is also a tendency to produce a longitudinal tilting of the disc, but as this term is only of the order of one tenth of the lateral moment and the pitching moment of inertia is about ten times that in roll, the longitudinal effect may be considered as negligible in comparison to the lateral effect. Stated in general and simple form, the effect of a fore-and-aft stick movement changes the attitude of the helicopter, while flying straight into a gust causes the helicopter to roll.

6. BLIND FLYING.

Now that helicopters have come into commercial use, one of the most important considerations from the point of view of successful operation is the ability to fly in practically any weather conditions, particularly in limited visibility and at night. Such operation may be divided into two main subjects; first, the ability to maintain satisfactory control of the helicopter using instruments and without reference to external visual aids and second, the navigational problems. Only the former is considered here.

The problem of blind flying from the pilot's viewpoint is to determine from the instruments the precise flight conditions of the helicopter. The technicians' aim is to provide the instruments which will perform this function, with a minimum of fatigue and concentration to the pilot. Before discussing the instruments in particular, it cannot be emphasised too strongly that ease of blind flying is fundamentally associated with satisfactory stability and control characteristics and, in particular, with the ability to trim out any forces in the pilot's controls.

The main difference in blind flying a helicopter compared with a fixed-wing aircraft, is in the longitudinal behaviour; laterally and directionally both types show very similar behaviour. For the helicopter with hinged rotor blades the fuselage attitude is simply a function of the forces acting on the fuselage. In steady flight, the fuselage tilt depends only on the drag and weight. The attitude therefore depends only on speed and is virtually independent of the rate of climb or descent, except at high angles of climb or descent where the inclination of the drag becomes important.

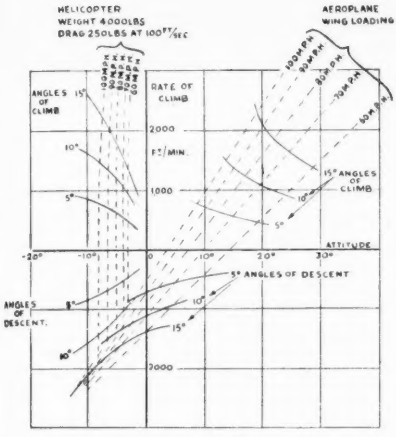


Fig. 15.
Comparison of attitude of helicopter and aeroplane.

Figure 15 gives a comparison of the helicopter attitude with that of a fixed-wing aircraft. A low wing loading has been assumed for the latter to give speeds comparable with those of the helicopter. A similar comparison could be made with a higher wing loading aeroplane, but the speeds would be so high that the comparison would have to be made generally, in terms of attitudes at top speed, cruising speed, climbing conditions and so on. The fundamental difference in the two types is shown by the lines of constant speed, where the helicopter attitude remains substantially constant for any rate of climb or descent while that of the fixed-wing aeroplane varies considerably. The comparison becomes quite striking if a simple case is considered. For example, the helicopter climbing 1,000 ft./min. at 60 m.p.h. (an 11 degree angle of climb) and then descending at 100 m.p.h. and 2,000 ft./min. has to change its attitude by about 5 degrees. On the fixed-wing aircraft the change is about 35 degrees. If the helicopter descent had been made at the same speed as on the climb, the change in attitude would have been negligible, while for the aeroplane it would still have been as much as 30 degrees.

It is immediately obvious that the standard artificial horizon is of no use for giving an absolute indication of the changes of attitude with speed or rate of climb. It does, however, respond normally to any change in attitude, *e.g.* due to atmospheric bumps and so on. If a more sensitive artificial horizon is used (greater movement of the bar for equivalent attitude change) the position of the bar gives a better indication of speed in the steady conditions but the movements for arbitrary disturbances are also exaggerated and this tends to cause over-corrections from the pilot. The artificial horizon, while it may be of some use as an interim measure, should not be considered as the final solution to the problem of giving the pilot a reliable indication of the longitudinal conditions.

Two minor points which must not be overlooked in the use of the standard artificial horizon are, first, the normal turning error of the instrument which must be reset to the lower speed of the helicopter from the standard value of 200 m.p.h. and second, the effect of the fuselage in a turn. In the latter, the drag of the fuselage remains the same for the same flight speed but the normal force has increased. The movement of the bar due to this effect, while very small on the standard horizon, was quite noticeable on the sensitive instrument.

The angle of bank indication on the artificial horizon is straightforward. Similarly, the behaviour of the "gyro direction indicator" is quite orthodox. The "turn and bank" operates correctly, but due to small pendular movements of the helicopter, the readings are much too sensitive, particularly the sideslip needle, and cannot be read accurately.

An aerodynamic means of sideslip indication (using the hemispherical type of head) was tried and proved entirely satisfactory. In view of the ability to prevent sideslip accurately in level flight by suitable co-ordination of the other instruments when a complete blind flying panel is used, it is doubtful if the additional complication of fitting such an instrument is justified. Nevertheless, if a more accurate knowledge of sideslip in turns is required than that given by the "turn and bank," some such sideslip indicator will be necessary.

It is early in the stage of helicopter blind flying development to provide definite conclusions, but it would appear that satisfactory blind flying can be done with existing instruments with a modified flying technique. The most serious defect at the moment is the inability to give the pilot a reliable method of relating the attitude of the helicopter to the speed, and some better form of longitudinal indication is required. The air speed indicator becomes more important and is considered as the master instrument for the

longitudinal control of the helicopter. The angle of bank indication of the artificial horizon and the gyro direction indicator remain the master instruments in the other planes.

Under these conditions, flying in cloud and "under the hood" and night flying have been done quite successfully on the Sikorsky R-4B, but considerable concentration is required from the pilot.

7. CONCLUDING REMARKS.

In this lecture I have dealt with the most important items in recent research flight testing of the helicopter at the Royal Aircraft Establishment. These are but a few items compared with the many problems of the helicopter which still remain to be tackled. In conclusion, I would like to summarise the results of the flight testing work on the helicopter to date, to show the present position in helicopter development and to give some indication of the problems still to be solved.

Perhaps the most important feature of the recent flight testing work is the satisfactory establishment of suitable techniques which can be applied in the various branches of aerodynamic research to the full-scale helicopter. These investigations have enabled the flight testing of helicopters to be rapidly founded on a sound scientific basis.

The work on the flow conditions through the rotor disc has established the application of propeller strip theory to the full-scale rotor. While the present work will be sufficient for the accurate evaluation of the induced power losses in performance estimation, further work may be necessary at higher speeds where the flow through the disc varies with azimuth angle and has an important influence on the stability and control characteristics. The difficulties in the experimental verification of these flow conditions at high forward speed will be very great. The small order of the required measurements, the rapid development of the slipstream, the inter-action of the thrust gradings at the various azimuth positions during the slipstream development, and the problems of determining the flight conditions with sufficient accuracy, are but a few of the difficulties that will be encountered.

It is in stability and control that the helicopter development is most backward. No one can claim that the flying characteristics of the helicopter are at the moment satisfactory. The development of a helicopter which can be flown with the same ease and lack of fatigue and concentration now associated with the fixed-wing aeroplane may take some time. Only the simultaneous advancement of the stability and control flight work with the theoretical treatment of the subject can show where these improvements are to be found.

The helicopter, as we know it at the moment, is completely unsatisfactory dynamically under stick *free* conditions and although conditions are improved with the stick *fixed*, the stability characteristics cannot be said to approach anywhere near to the desired standard. The vital need is for some means of accurately trimming out the forces in the pilot's stick. This, and the divergent movement of the stick when it is freed, have led to the fitting of irreversible controls on recent American types. This also prevents vibration in the rotor mechanism being transmitted back to the pilot's stick. While it may be of considerable assistance to the pilot at the present stage of helicopter development, it has several disadvantages and may not be the final solution to the unsatisfactory stick-free behaviour. The ability to fly the helicopter on instruments in poor visibility, or at night, makes even higher demands for the improvement of the stability and control characteristics.

The development of suitable aerofoil sections for helicopter blades is a problem which has, as yet, received little attention. A great many conflicting conditions must be satisfied for such an aerofoil. If the forward speed of the helicopter is to be increased, the tip speed of the blades must be increased as far as compressibility effects will allow. The compressibility troubles apply particularly at the tip of the advancing blade where the rotational speed and forward flight speed are additive, but may also be important for the retreating blade where the Mach number is lower, but the working lift coefficient is much higher. Consideration must be given to obtaining as high a maximum lift coefficient as possible to delay the stalling of the retreating blade. The possibility that the designer may have to resort to flaps or slots on the blades must not be overlooked. Another important consideration is that the aerofoil should have zero pitching moment and that there should be no change in the pitching moment with incidence. This is desirable at the moment, but may have to be modified when more experience has been obtained with the development of satisfactory stick-free behaviour in the helicopter.

In evolving an aerofoil to meet these conditions, it is important that its characteristics should not be affected by small changes in the section. Such distortion of the section may take place in production, but a more likely source is due to twisting or bending of the blade in flight. This difficulty may apply to the recent N.A.C.A. sections suggested for helicopter use. These sections depend on the action of the reflex cambered trailing edge to give the zero C_{mo} and to maintain no change of C_m with incidence. Slight distortions of the trailing edge, which can so easily occur in flight with present methods of blade construction, affect the reflex camber and entirely change the properties of the aerofoil section.

The development of suitable blade sections, especially with reference to the high maximum lift coefficient, is becoming increasingly important. We are now reaching the stage where, if the helicopter is designed to have a positive rate of climb at zero air speed and if care is taken in the fuselage design, the top speed of the helicopter will not be limited by power considerations but by the excessive flapping and serious vibration due to the transient blade stalling. This stalling of the retreating blade must be considered from two aspects; first, the large increase in drag due to the stalling and, second, the effect on the handling of the helicopter. While it is true to say that the performance of the helicopter is affected right from the initial onset of the stalling, it is possible to exceed these conditions by a considerable amount before the effects on the helicopter behaviour become serious. Some excellent work on this subject has already been done in America, but as we are only beginning to experience the stalling limitations to helicopter flight, this work is also only in its initial stages. This is, of course, in addition to considerations of delaying the stall by twisting of the blade, or by the use of some alternative configuration, such as the gyrodyne.

There are still many more problems to be solved not only in the aerodynamic field, but in structural design, particularly in the vibration work. The advent of the multi-rotor configuration and the application of jet propulsion to helicopters bring with them their own particular problems.

The next few years will be of great interest in helicopter work and should bring rapid advances in helicopter development. Much ingenuity in design will be required to surmount the many problems involved, but the final success or otherwise of any design is proved in the flight testing. Only the simultaneous advancement of design and theoretical work with careful testing, in flight and on models, can give success to the development of the helicopter.

8. ACKNOWLEDGMENTS.

I would like to thank my colleagues at the Royal Aircraft Establishment, particularly Messrs. Brotherhood and Zbrozek of the helicopter flight group, for the assistance given during numerous discussions on the many and varied problems of the helicopter research work. I would also like to include the experimental test pilots at Farnborough, particularly Lt./Cdr. E. M. Brown who did most of the piloting for the work described in this lecture and assisted in the development of the many novel techniques required in these investigations.

I also wish to express my thanks to the Ministry of Supply for permission to publish this paper.

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LIST OF SYMBOLS

A		Non-dimensional coefficient of rotor
a	C_L/rad.	Slope of lift curve of blade section
a_0	rad.	Coning angle
a_1	rad.	Coefficient of $\cos\psi$ in Fourier's series for flapping angle β
a_{1_u}		$(\partial a_1/\partial u)$
a_{1_q}		$(\partial a_1/\partial q)$
$a_{1_{\mu_1}}$		$(\partial a_1/\partial \mu_1)$
a_{1_α}		$(\partial a_1/\partial \alpha)$
B		Factor allowing for tip loss; generally taken as 0.98
b		Number of blades of rotor
b_1	rad.	Coefficient of $\sin\psi$ in Fourier's series for flapping angle
c	ft.	Chord of blade at $0.7R$
C_L, C_D		Lift, drag coefficients of rotor referred to flight speed and disc area $C_L = L/(\frac{1}{2}\rho V^2 \pi R^2) = (kL/\mu_1{}^2)$ $C_D = D/(\frac{1}{2}\rho V^2 \pi R^2) = (kD/\mu_1{}^2)$
C_{D_f}		Fuselage drag coefficient $C_{D_f} = Df/(\frac{1}{2}\rho V^2 \pi R^2)$
C_{m_f}		Fuselage pitching moment coefficient $C_{m_f} = Mf/(\frac{1}{2}\rho V^2 \pi R^2 \,.\, R)$
C_{L_α}		$(\partial C_L/\partial \alpha)$
$C_{L_{\mu_1}}$		$(\partial C_L/\partial \mu_1)$
C_{D_α}		$(\partial C_D/\partial \alpha)$
$C_{D_{\mu_1}}$		$(\partial C_D/\partial \mu_1)$
C_{Dfi}		$(\partial C_{Df}/\partial i)$
D	lb.	Rotor drag
D_f	lb.	Fuselage drag
fR	ft.	Fore and aft position of rotor shaft relative to c.g. of helicopter
f_1		$f\cos i - h\sin i$
f		Suffix denoting fuselage
hR	ft.	Distance of rotor hub above c.g.
h_1		$h\cos i + f\sin i$
I	ft. lb./sec.^{2}	Total moment of inertia of helicopter about y-axis
I_b	ft. lb./sec.^{2}	Contribution of the blades to I where $I_b = \{ bW_b (hR)^2/g \} \; (1 + f^2/h^2 + a_0 s/h)$
I_1	ft. lb./sec.^{2}	Moment of inertia of blade about flapping hinge
i	rad.	Rotor incidence, angle between relative wind and a plane perpendicular to the rotor shaft
k_L, k_D		Lift, drag coefficients of rotor referred to tip speed and disc area $k_L = L/\{\frac{1}{2}\rho(\Omega R)^2 \pi R^2\}$ $k_D = D/\{\frac{1}{2}\rho(\Omega R)^2 \pi R^2\}$
k_T		Thrust coefficient $k_T = T/\{\frac{1}{2}\rho(\Omega R)^2 \pi R^2\}$
k_{L_α}		$(\partial k_L/\partial \alpha)$
$k_{L_{\mu_1}}$		$(\partial k_L/\partial \mu_1)$
k_{D_α}		$(\partial k_D/\partial \alpha)$
$k_{D_{\mu_1}}$		$(\partial k_D/\partial \mu_1)$
L	lb.	Rotor lift

M	lb. ft.	Pitching moment
M_1	lb. ft.	Weight moment of blade about flapping hinge
M_u		$(\partial M/\partial u)$
M_w		$(\partial M/\partial w)$
M_q		$(\partial M/\partial q)$
q	rad./sec.	Angular velocity of helicopter about the y-axis
q_0	rad./sec.	Angular velocity of helicopter about the rotor hub centre
r		Root of the frequency equation
r		Suffix denoting rotor
R	ft.	Rotor radius
πR^2	ft.^{2}	Rotor disc area
sR	ft.	Distance of c.g. of blade from flapping hinge
T	lb.	Thrust of rotor in hovering flight
T_0	sec.	Periodic time of oscillation
T_H	sec.	Time to half amplitude in a damped oscillation
T_D	sec.	Time to double amplitude in an increasing oscillation
u	ft./sec.	Small disturbance of velocity V in the x-direction
V	ft./sec.	Velocity of undisturbed flight
w	ft./sec.	Small disturbance of velocity V in the z-direction
W	lb.	Weight of helicopter
W_b	lb.	Weight of one rotor blade
x, y, z		Rectangular system of axes with origin coincident with c.g. of helicopter and the x-axis indicates direction of undisturbed flight
X, Y, Z	lb.	Forces in direction of corresponding axes
X_u, X_w, X_q		$\partial X/\partial u$; $\partial X/\partial w$; $\partial X/\partial q$
Z_u, Z_w, Z_q		$\partial Z/\partial u$; $\partial Z/\partial w$; $\partial Z/\partial q$
α	rad.	Effective angle of attack of rotor $= i + \mathcal{E}s$
β	rad.	Flapping angle $\beta = a_0 - a_1 \cos\psi - b_1 \sin\psi \ldots$
γ_0		*Mass constant of blade $= (R^4 \rho c a)/I_1$
γ		Apparent mass constant of blade
η		Non-dimensional aerodynamic coefficient of rotor
θ	rad.	Angle of rotation in pitch
$\theta_1; \theta_2$		Angles of rotation in pitch at times t_1 and t_2
$\mathcal{E}$	rad.	Instantaneous pitch of blade at 0.7 rad. and given by $\mathcal{E}_0 + \mathcal{E}_s \sin\psi$
$\mathcal{E}_0$	rad.	Pitch of blade with no cyclical pitch imposed
$\mathcal{E}_s$	rad.	Cyclical pitch due to longitudinal control
λ		Damping coefficient for the longitudinal motion
μ_1		Tip speed ratio $V/\Omega R$
$\mu_{1\alpha}$		$(\partial\mu_1/\partial\alpha)$
ν		Non-dimensional aerodynamic coefficient of rotor
ρ	slugs/ft.^{3}	Air density
σ		Solidity of rotor (blade area/disc area) $= (bc/\pi R^2)$
τ	rad.	Angle of climb
χ	rad.	Attitude of rotor shaft relative to the vertical positive if the shaft is inclined backward

* The mass constant γ_0 is defined more fully after equation (17).

ψ	rad.	Blade azimuth angle measured from downwind position in the direction of rotation
Ω	rad./sec.	Angular velocity of rotor
ΩR	ft./sec.	Tip speed of rotor

APPENDIX I

Calculation of the longitudinal dynamic stability

In the following treatment of the theoretical estimation of the longitudinal stability, based on the work of G. J. Sissingh, only the particular case of the Sikorsky R-4B under the conditions of the flight tests is considered. In evaluating the various derivatives, the flight measurements of the variations of collective pitch, cyclical pitch, angle of tilt, rotor disc incidence and so on, with air speed have been used. The rotor derivatives in forward flight, for power-on conditions and in auto-rotation, are deduced from wind tunnel tests of inter-meshing rotors (ref. 6). The investigation of vertical descent in auto-rotation is based on model tests of autogyro rotors (ref. 4), and corrected for change in solidity.

The available data on the pitching moments of the fuselage was poor. From measured values at zero and high speed a linear relationship was assumed in the form

$$C_{mf}=(-0.572+2.14i)\,10^{-3}$$

Equations of motion

The equations of motion are referred to the normal rectangular x, y, z system of axes, with the origin coincident with the centre of gravity of the helicopter and the x direction is parallel to the flight path.

The equations for the equilibrium of the forces and moments (see Fig. 16) may be written

$$X_u u+X_w w+X_q q-W\theta\cos\tau-(W/g)\,\dot{u}=0 \quad . \quad . \quad . \quad (1)$$

$$Z_u u+Z_w w+Z_q q-W\theta\sin\tau+(W/g)\,Vq-(W/g)\,\dot{w}=0 \quad . \quad (2)$$

$$M_u u+M_w w+M_q q-I\dot{q}=0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (3)$$

The pitching moment about the centre of gravity due to the rotor (see Fig. 17) is given as

$$M_r=-RX_r\,(h\cos i+f\sin i)$$
$$-RZ_r\,(f\cos i-h\sin i) \quad . \quad . \quad . \quad . \quad . \quad . \quad (4)$$

Substituting $h_1=h\cos i+f\sin i \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (5)$

and $f_1=f\cos i-h\sin i \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (6)$

Equation (4) gives

$$M_{ru}=-RX_{ru}h_1-RZ_{ru}f_1 \quad . \quad . \quad . \quad . \quad . \quad (7)$$

$$M_{rw}=-RX_{rw}h_1-RZ_{rw}f_1 \quad . \quad . \quad . \quad . \quad . \quad (8)$$

$$M_{rq}=-RX_{rq}h_1-RZ_{rq}f_1 \quad . \quad . \quad . \quad . \quad . \quad (9)$$

The rotation of the helicopter about its centre of gravity imposes linear velocities on the hub

$-h_1Rq$ in the x direction

and $-f_1Rq$ in the z direction

Hence we may write

$$X_{rq}=X_{rqo}-h_1RX_{ru}-f_1RX_{rw} \quad . \quad . \quad . \quad . \quad . \quad (10)$$

$$Z_{rq}=Z_{rqo}-h_1RZ_{ru}-f_1RZ_{rw} \quad . \quad . \quad . \quad . \quad . \quad (11)$$

$$a_{1q}=a_{1qo}-\frac{h_1}{\Omega}a_{1_{\mu_1}}-\frac{f_1}{\Omega\mu_1}a_{1_\alpha} \quad . \quad . \quad . \quad . \quad (12)$$

In these equations X_{rqo}, Z_{rqo}, $a_{1_{\mu_1}}$ and a_{1_α} represent rates of change with respect to the angular velocity about the hub centre.

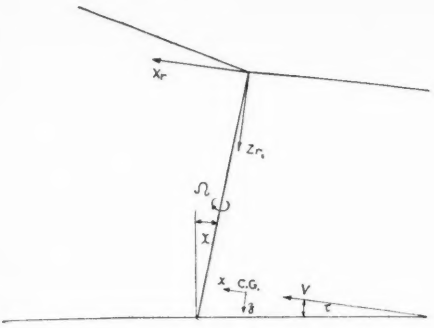

Fig. 16.
Nomenclature diagram.

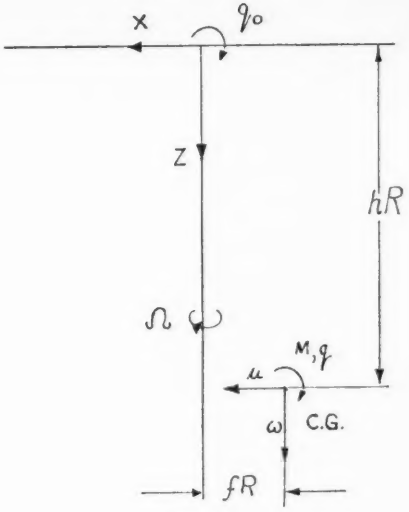

Fig. 18.
Hovering flight.

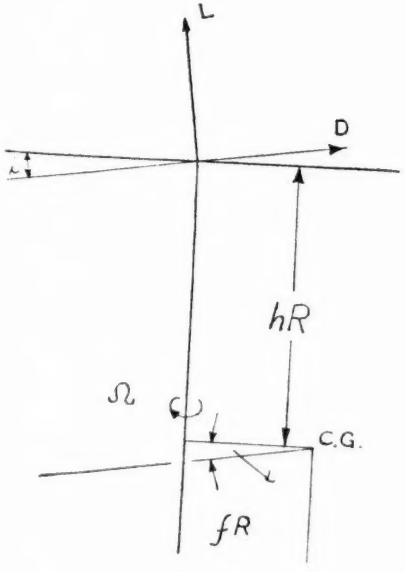

Fig. 17.
Nomenclature diagram.

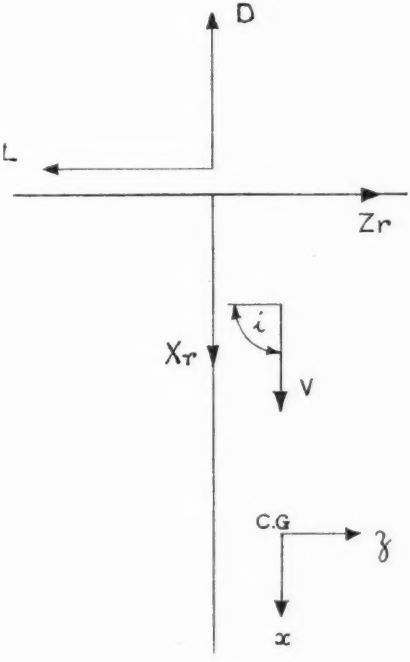

Fig. 19.
Vertical descent.

The derivatives X_{ru}, X_{rw}, Z_{ru}, Z_{rw}, $a_{1_{\mu_1}}$ and a_{1_α}, are generally obtained from wind tunnel data, *i.e.* from lift and drag curves against speed ratio and "effective" rotor incidence α where the latter is given by

$$\alpha = i + \ell_s \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (13)$$

$$= \chi - \tau + \ell_s \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (14)$$

There are no available measurements of the rotary derivatives and these must be estimated from the following equations

$$a_{1qo} = -16/\{\gamma\Omega(B^4 - \tfrac{1}{2}B^2\cos^2 i)\} \quad . \quad . \quad . \quad . \quad (15)$$

$$X_{rqo} = -\eta T a_{1qo} = 16\eta_1 W/\{\gamma\Omega(B^4 - \tfrac{1}{2}B^2\cos^2 i)\} \quad . \quad . \quad (16)$$

$$Z_{rqo} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (17)$$

The mass constant γ_o, representing the relationship of the air forces to the mass forces, is obtained by considering the moments of these forces about the flapping hinge, and the air forces involved are based on the assumption of a constant flow induced velocity field. In practice, the induced velocity distribution may vary considerably from this assumption and the tip and root losses must be taken into consideration. Calculations based on these latter conditions give the value of the apparent mass constant γ and this value may be considerably less than γ_o at very low tip speed ratios, where the effect of the induced velocity distribution is most important.

The coefficients η and γ/γ_o depend on the nature of the flow through the disc and can be expressed in terms of collective pitch ℓ_o thrust coefficient k_T and solidity σ. Both factors increase with increase in tip speed ratio and approach the value unity.

HOVERING FLIGHT.

Hovering flight may be interpreted as a level flight condition with velocity $V = 0$ and the x axis horizontal, *i.e.* $\tau = 0$. These conditions are shown diagrammatically in Fig. 18. As the influence of the fuselage derivatives may be neglected and the rotor derivatives X_{rw}, Z_{rq} and M_{rw} are zero, the equations of motion become

$$X_{ru}u + X_{rq}q - W\theta - (W/g)\dot{u} = 0 \quad . \quad . \quad . \quad . \quad . \quad (18)$$

$$Z_{rw}w - (W/g)\dot{w} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad (19)$$

$$M_{ru}u + M_{rq}q - I\dot{q} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad (20)$$

From equation (19) it will be seen that the vertical motion of the helicopter is aperiodic. The exponential index (r) is $g(Z_{rw}/W)$ and as Z_{rw} is always negative, this disturbance is damped. Equations (18) and (20) give the frequency equation

$$r^3(WI/g) - r^2[(M_{rq}W/g) + IX_{ru}] + r(X_{ru}M_{rq} - X_{rq}M_{ru}) + WM_{ru} = 0 \quad . \quad (21)$$

For the Sikorsky configuration, f is negligible compared with h (for the centre of gravity on the rotor shaft $f = 0$).

Hence,

$$M_{rq} = -hRX_{rq} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (22)$$

$$M_{ru} = -hRX_{ru} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (23)$$

$$M_{rw} = -hRX_{rw} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (24)$$

Thus, the term r in the equation (21) vanishes and by Routh's criterion, hovering flight instability with this configuration is unavoidable.

If no wind tunnel data is available, the necessary rotor derivatives may be calculated from the following equations

$$X_{ru} = WA/\Omega R \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (25)$$

$$X_{rw} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (26)$$

$$X_{rqo} = (16\eta W/\gamma\Omega B^4) \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (27)$$

$$X_{rq} = (W/\Omega)\{(16\eta/\gamma B^4) + Ah\} \quad . \quad . \quad . \quad . \quad . \quad . \quad (28)$$

$$Z_{ru} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (29)$$

$$Z_{rw} = -(W/\Omega R)\{2/(2-\eta)\sqrt{k_T}\} \quad . \quad . \quad . \quad . \quad . \quad . \quad (30)$$

$$Z_{rqo} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (31)$$

$$Z_{rq} = (W/\Omega R)\{2f/(2-\eta)\sqrt{k_T}\} = 0 \text{ for } f = 0 \quad . \quad . \quad . \quad (32)$$

M_{rq}, M_{ru}, M_{rw} are as given in (22), (23), (24).

HELICOPTER FORWARD FLIGHT.

The lift and drag coefficients of the helicopter rotor depend on the incidence α and the tip speed ratio μ_1. The coefficients C_L and C_D are referred to the disc area and forward velocity while k_L and k_D are referred to disc area and tip speed.

For equilibrium

$$k_L = (W \cos \tau) / \{ \tfrac{1}{2}\rho (\Omega R)^2 \pi R^2 \} \quad . \quad . \quad . \quad . \quad (33)$$

$$k_D = - [C_{D_f} \mu_1^2 + W \sin \tau / \{ \tfrac{1}{2}\rho (\Omega R)^2 \pi R^2 \}] \quad . \quad . \quad (34)$$

Assuming that the rotor speed is not influenced by the small disturbances, the changes in incidence and tip speed ratio are given by

$$\Delta\alpha = (w/V) \quad . \quad . \quad . \quad . \quad . \quad . \quad (35)$$

$$\Delta\mu_1 = (u/\Omega R) \quad . \quad . \quad . \quad . \quad . \quad . \quad (36)$$

Hence,

$$\Delta L = \tfrac{1}{2}\rho (\Omega R)^2 \pi R^2 [k_{L_\alpha} (w/V) + k_{L_{\mu_1}} (u/\Omega R)] \quad . \quad . \quad . \quad (37)$$

$$\Delta D = \tfrac{1}{2}\rho (\Omega R)^2 \pi R^2 [k_{D_\alpha} (w/V) + k_{D_{\mu_1}} (u/\Omega R)] \quad . \quad . \quad . \quad (38)$$

The rotor force may be written

$$\Delta X_r = L\Delta\alpha - \Delta D \quad . \quad . \quad . \quad . \quad . \quad . \quad (39)$$

$$\Delta Z_r = - (\Delta L + D\Delta\alpha) \quad . \quad . \quad . \quad . \quad . \quad (40)$$

and from (37) and (38)

$$\Delta X_r = \tfrac{1}{2}\rho (\Omega R)^2 \pi R^2 [k_{L_\alpha} (w/V) - k_{D_\alpha} (w/V) - k_{D_{\mu_1}} (u/\Omega R)] \quad . \quad . \quad (41)$$

$$\Delta Z_r = - \tfrac{1}{2}\rho (\Omega R)^2 \pi R^2 [k_{L_{\mu_1}} (w/V) + k_{L_{\mu_1}} (u/\Omega R) + k_D (w/V)] \quad . \quad . \quad (42)$$

Hence, we obtain the force derivatives,

$$X_{ru} = - \tfrac{1}{2}\rho (\Omega R) \pi R^2 k_{D_{\mu_1}} \quad . \quad . \quad . \quad . \quad . \quad . \quad (43)$$

$$X_{rw} = \tfrac{1}{2}\rho (\Omega R) \pi R^2 (1/\mu_1) (k_L - k_{D_\alpha}) \quad . \quad . \quad . \quad . \quad . \quad (44)$$

$$Z_{ru} = - \tfrac{1}{2}\rho (\Omega R) \pi R^2 k_{L_{\mu_1}} \quad . \quad . \quad . \quad . \quad . \quad . \quad (45)$$

$$Z_{rw} = - \tfrac{1}{2}\rho (\Omega R) \pi R^2 (1/\mu_1) (k_D + k_{L_\alpha}) \quad . \quad . \quad . \quad . \quad . \quad (46)$$

The values of k_{L_α}, $k_{L_{\mu_1}}$, k_{D_α} and $k_{D_{\mu_1}}$ are generally obtained from wind tunnel tests. If measurements of the drag derivatives are not available, approximate values can be deduced in terms of the lift and flapping angle, from the relationship

$$D = L (\alpha + a_1) \quad . \quad . \quad . \quad . \quad . \quad . \quad (47)$$

From this, equations (43) and (44) may be written in the form

$$X_{ru} = - \tfrac{1}{2}\rho (\Omega R) \pi R^2 [k_L a_{1_{\mu_1}} + (k_D/k_L) k_{L_{\mu_1}}] \quad . \quad . \quad . \quad (48)$$

$$X_{rw} = - \tfrac{1}{2}\rho (\Omega R) \pi R^2 (1/\mu_1) [k_L a_{1_\alpha} + (k_D/k_L) k_{L_\alpha}] \quad . \quad . \quad (49)$$

If the lift and drag coefficients of the rotor are referred to the forward speed, equations (43) (44), (45), (46), (48) and (49) take the form

$$X_{ru} = - \tfrac{1}{2}\rho V \pi R^2 (2C_D + \mu_1 C_{D_{\mu_1}}) \quad . \quad . \quad . \quad . \quad . \quad (43a)$$

$$X_{rw} = + \tfrac{1}{2}\rho V \pi R^2 (C_L - C_{D_\alpha}) \quad . \quad . \quad . \quad . \quad . \quad (44a)$$

$$Z_{ru} = - \tfrac{1}{2}\rho V \pi R^2 (2C_L + \mu_1 C_{L_{\mu_1}}) \quad . \quad . \quad . \quad . \quad . \quad (45a)$$

$$Z_{rw} = - \tfrac{1}{2}\rho V \pi R^2 (C_D + C_{L_\alpha}) \quad . \quad . \quad . \quad . \quad . \quad (46a)$$

$$X_{ru} = - \tfrac{1}{2}\rho V \pi R^2 [2C_D + \mu_1 (C_D/C_L) C_{L_{\mu_1}} + \mu_1 C_L a_{1_{\mu_1}}] \quad . \quad (48a)$$

$$X_{rw} = - \tfrac{1}{2}\rho V \pi R^2 [C_L a_{1_\alpha} + (C_D/C_L) C_{L_\alpha}] \quad . \quad . \quad . \quad (49a)$$

Considering the influence of the fuselage and denoting the drag and pitching moment coefficients by

$$C_D = D_f / (\tfrac{1}{2}\rho V^2 \pi R^2) \quad . \quad . \quad . \quad . \quad . \quad . \quad (50)$$

$$C_m = M_f / (\tfrac{1}{2}\rho V^2 \pi R^2 . R) \quad . \quad . \quad . \quad . \quad . \quad . \quad (51)$$

the fuselage derivatives may be written

$$X_{f_u} = - \tfrac{1}{2}\rho V \pi R^2 2C_{D_f} \quad . \quad . \quad . \quad . \quad . \quad . \quad (52)$$

$$X_{fw} = - \tfrac{1}{2}\rho V \pi R^2 C_{D_{fl}} \quad . \quad . \quad . \quad . \quad . \quad . \quad (53)$$

$$Z_{fu} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (54)$$

$$Z_{fw} = - \tfrac{1}{2}\rho V \pi R^2 C_{D_f} \quad . \quad . \quad . \quad . \quad . \quad . \quad (55)$$

$$M_{fu} = \tfrac{1}{2}\rho V \pi R^2 . R 2C_{m_f} \quad . \quad . \quad . \quad . \quad . \quad . \quad (56)$$

$$M_{fw} = \tfrac{1}{2}\rho V \pi R^2 . R C_{m_{fl}} \quad . \quad . \quad . \quad . \quad . \quad . \quad (57)$$

AUTO-ROTATION.

In auto-rotational flight, the incidence and tip speed ratio of the rotor are no longer independent of each other and the rotational speed of the rotor must be considered as an additional degree of freedom. It is difficult to deal with the changes of rotor rotational speed with incidence during the motion and it is assumed that at each instant the rotational speed corresponds to that of the steady state for the given incidence. The variation of rotor speed with incidence for the required speed range was measured in flight. The variation of the rotational speed of the blade during its azimuth travel is also neglected.

In steady flight,

$$C_L = (W \cos \tau)/(\tfrac{1}{2}\rho V^2 \pi R^2) \quad . \quad . \quad . \quad . \quad . \quad (58)$$

$$C_D = -C_{D_f} + (W \sin \tau)/(\tfrac{1}{2}\rho V^2 \pi R^2) \quad . \quad . \quad . \quad (59)$$

Hence, the force derivatives of the rotor are given by

$$X_{ru} = -\tfrac{1}{2}\rho V \pi R^2 2 C_D \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (60)$$

$$X_{rw} = +\tfrac{1}{2}\rho V \pi R^2 (C_L - C_{D_\alpha}) \quad . \quad . \quad . \quad . \quad . \quad (61)$$

$$Z_{ru} = -\tfrac{1}{2}\rho V \pi R^2 2 C_L \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (62)$$

$$Z_{rw} = -\tfrac{1}{2}\rho V \pi R^2 (C_D - C_{L_\alpha}) \quad . \quad . \quad . \quad . \quad . \quad (63)$$

Numerical values of C_{L_α} and C_{D_α} are generally obtained from wind tunnel tests. If the curves of C_L and C_D are plotted against tip speed ratio, it is convenient to write (61) and (63) in the form

$$X_{rw} = +\tfrac{1}{2}\rho V \pi R^2 (C_L - C_{D_{\mu_1}} \mu_{1_\alpha}) \quad . \quad . \quad . \quad . \quad . \quad (64)$$

$$Z_{rw} = -\tfrac{1}{2}\rho V \pi R^2 (C_D + C_{L_{\mu_1}} \mu_1) \quad . \quad . \quad . \quad . \quad . \quad (65)$$

If the drag has to be estimated from the lift and flapping motion, (61) takes the form

$$X_{rw} = -\tfrac{1}{2}\rho V \pi R^2 \{ (C_D/C_L) C_{L_\alpha} + C_L a_{1_\alpha} \} \quad . \quad . \quad . \quad . \quad (66)$$

or

$$X_{rw} = -\tfrac{1}{2}\rho V \pi R^2 \{ (C_D/C_L) C_{L_{\mu_1}} \mu_{1_\alpha} + C_L a_{1_{\mu_1}} \mu_1 \} \quad . \quad . \quad . \quad (67)$$

VERTICAL DESCENT IN AUTO-ROTATION.

In vertical descent, Fig. 19,

$$\tau = -90°, \; C_L = 0, \; C_D \doteq W/(\tfrac{1}{2}\rho V^2 \pi R^2)$$

Also, if the rotor shaft is vertical and no cyclical pitch is imposed on the blades.

$$\chi = 0 \text{ and } \alpha = 90°$$

and due to symmetry

$$C_{D_\alpha} = 0.$$

Neglecting the force derivatives of the fuselage we have

$$X_w = Z_u = 0$$

and the equations of motion become

$$X_u u - (W/g)\dot{u} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (68)$$

$$Z_w w + Z_q q + W\theta + (W/g) V q - (W/g)\dot{w} = 0 \quad . \quad . \quad . \quad . \quad (69)$$

$$M_w w + M_q q - I\dot{q} = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad (70)$$

As in the hovering state, the vertical motion is no longer coupled with the longitudinal and pitching motions of the helicopter. The vertical motion dies out aperiodically and the exponential index is given by $(X_u g/W)$.

The other two equations give the frequency equation

$$r^3 (WI/g) - r^2 [Z_w I + (W/g) M_q] + r [Z_w M_q - M_w Z_q - M_w (W/g) V] - M_w W = 0 \quad . \quad (71)$$

and the term $(Z_w M_q - M_w Z_q)$ is again zero.

The equations for the estimation of the derivatives are as before. In determining Z_{rw}, it should be noted that C_{L_α} is negative in this condition.

DISCUSSION

N. E. Rowe (British European Airways, *Fellow*): Mr. Stewart had referred to some of the air flow work being done by attaching a smoke canister to the end of the jib of a 60 ft. crane, with the helicopter hovering underneath at a height of 40 ft. That seemed to involve something tricky in the way of piloting, calling for care and skill of a high order. The work of establishing the vortex ring pictures involved piloting of a difficult nature, and he would imagine, dangerous.

It seemed to him that it should be possible to obtain a great deal of information, of a quantitative kind as well as of the qualitative kind shown by Mr. Stewart, by work in the tunnel. What were the prospects and was there any programme for such tunnel work?

Comment had rightly been made on the great importance of stability in blind flying and on the lack of stability of the single-rotor helicopters they were flying at the moment. If they were to use helicopters in these islands they must be able to fly under all conditions of visibility; in order to do that in safety they must have machines of adequate stability. In that connection he felt that the application of tunnel technique to the different kinds of configurations which were now coming along, including perhaps the establishing of flow pictures by the smoke technique, might be valuable, and he imagined the work could be done much more quickly and much more safely in that way than by full-scale testing in the air.

He urged that the helicopter should be treated on its merits in blind flying and they should not force into use the same blind flying instruments, even if they were arranged in a different way, as were used in fixed-wing machines for blind flying. The conditions for the two types of aircraft were fundamentally different. The main risk against which they must safeguard in the case of the fixed-wing machine was loss of control at the stall; in the helicopter there was no such danger, and the characteristics of the aircraft ought to be considered in relation to blind flying in a fundamental way so that a basic appreciation of what they wanted to do was obtained. Perhaps they would have to modify that slightly to deal with the unstable aircraft that were available at present, but they had to consider those of the future.

His personal conviction was that this country offered an almost unrivalled field for the operation of that class of aircraft. They had urban agglomerations in various parts of their islands, and if these were to be served by air at all, the helicopter was the machine to provide that service. Much would depend on the early success of various models that were building in this country now, but in view of what he regarded as the certainty of the arrival of the helicopter, and hence the importance of giving the designers adequate information, he pleaded for a much more active research programme because without it progress was bound to be slow.

R. Hafner (Bristol Aeroplane Co. Ltd.): Mr. Stewart had referred to his suggested curve for induced flow in the vortex flow region; that curve was based on some 20 per cent. theory and 20 per cent. experimental data, the remainder being made up by pure guess-work. Therefore, he was quite prepared to scrap that curve wherever it was in disagreement with Mr. Stewart's findings. The vortex region was characterised by intense vorticity and considerable mixing of air. That was a feature which apart from air inertia was bound up with viscous stresses. Therefore they must expect that the flow in the vortex region was subject to scale effect. Indeed, they knew that the drag of a solid disc, or that of an auto-rotating rotor, depended on Reynolds numbers, so that they might have there not only one characteristic curve, but a family of curves.

It had been stated that the helicopter was very unstable in pitch, particularly with the control column free. That feature had interested him for some time. Cyclic feathering involved inertia forces, as the blade had a certain moment of inertia about its pitch changing axis. Those inertia forces were transmitted through the control circuits to the control column and resulted in an increasing force with increase of speed, and that force was an unstable one, *i.e.*, it tended to push the control column forward as the speed increased. It had been shown by Mr. Stewart that one way of overcoming that force was to use irreversibility in the control system, as was done in some American systems. Another way was to employ elasticity; it could be shown how elasticity could counteract the inertia force, and in particular, in certain circumstances a torsion bar such as

the tie rod in the Bristol helicopter could be used to do the job. He felt that a combination of the tie rod with some drag in the system might be the answer to the problem.

Another matter of interest to him was that of aerofoils for rotor blades. In the design of the Bristol helicopter type 171 a reflex section was used, the reason being to obtain a high lift coefficient for the case of the retreating blade, to prevent stalling and at the same time, to ensure a zero C_{mo}. This coefficient being really zero over a wide range of incidences was very important from the point of view of flutter, as well as from the point of view of control forces. The reflex section met this requirement but was considered to be less satisfactory at high Mach numbers, and he was inclined now to go to a symmetrical section. Apart from certain manufacturing difficulties the reflexed aerofoil had shown up quite well in the Bristol 171. A Mach number of .75 had been reached at the advancing blade tip as a combination of a rotor speed of 263 r.p.m. and a translational speed of 120 m.p.h. without serious vibrations in the airframe or in the controls.

Lt./Cdr. (A) E. M. Brown (Royal Aircraft Establishment, *Associate*): As a pilot, in undertaking the flight testing he had met one or two difficulties which he thought could be erased by making them known to all and sundry. One of the difficulties had arisen because reports had been spread around to the effect that helicopters were difficult to fly, and in consequence the opportunities had not been given for pilots to fly the machines to the extent that one would like, without first of all undergoing a prolonged training course. That difficulty had been overcome now and there were one or two pilots with him at the meeting who were making a very fine show.

The helicopter was not a difficult machine to fly; indeed, he considered that it could easily be conquered within five hours. He had heard periods as long as 80 hours quoted; but a trouble they were up against was that the helicopter was a new machine, and it was felt that connected with it there were mysteries.

The flight testing did not present special problems; a test pilot well versed in the ordinary helicopter flying technique could easily cope with them fully. There had been one or two little problems, such as those of asymmetric trim loads, due to the big poles which Mr. Stewart had had sticking out of the undercarriage, and because he would have the c.g. moving about; but from time to time Mr. Stewart had relieved the physically-exhausted pilot and taken charge of the controls admirably, so that he was able to speak with more knowledge than that of a pure technician.

The problem of blind flying had been attacked at the R.A.E. only during the later stages of the flight testing, and the work had not been very intensive so far, but what had been done was revealing. Again, there had been talk of the difficulties of the blind flying of helicopters. In his view the difficulties were mainly, or at any rate to a large extent, psychological. It had been said, he believed erroneously, that the helicopter could not be stalled; in the course of his experience the helicopter had been put into some very precarious attitudes, even gyrating down-hill completely out of control. There was more than met the eye in the blind flying problem, but he believed it would prove to be one of the simpler problems to solve. If they could solve the psychological aspect of the problem they would solve 50 per cent. of the blind flying difficulties.

H. B. Squire (Royal Aircraft Establishment, *Fellow*): He agreed that a tailplane would perhaps remove all the difficulties in regard to the stability at cruising and at high forward speeds in the single-rotor helicopter. Then they would be left with the difficulties encountered in the hovering condition, which were not reduced by the tailplane; but, after all, the helicopter did not hover for a large part of its total flying time and there was a good deal to be said for the fitting of tailplanes to all new designs. He believed that designers were inclined that way.

He felt that the attempt to provide stick-free stability was likely to lead to difficulies, because attention would have to be given to details of blade attachment, aerodynamic forces on the blades, friction of the linkage system, and so on. It seemed better to abandon the attempt to provide stick-free stability, and either to provide irreversible control or to make the stick forces very light and to fit springs for trimming and to provide artificial "feel."

A programme for wind tunnel tests as suggested by Mr. Rowe was formally approved about four years ago, and eighteen months ago the actual design work on the

models was started; it was hoped to start the tests some time this year.

J. S. Glass: The special research programme at the R.A.E. was of the greatest assistance to the Airborne Forces Experimental Establishment in developing their methods of testing prototype helicopters. They had run into some of the problems mentioned in the paper concerning the measurement of low air speed. Fairly early in 1945 they had started using a vertical string to indicate zero air speed; but when testing the S.51 that method could not be used, because the pilot was sitting in the middle of the fuselage and could not see downwards; he could not stick his head out of the window. He believed that in the United States they had used an opposed-pitot type of yaw meter in the vertical plane to indicate when there was no forward speed; had Mr. Stewart any comments on that?

There had been produced at the A.F.E.E. a schedule of tests of prototype helicopters, which included brief measurements of stick forces. It seemed that there was a great need for a really quantitative assessment of aircraft stability from the testing point of view, and that a basic theory of helicopter stability would be of the greatest value not only to the test team, but also to the designers. He hoped the research programme at the R.A.E. would include the development of, or at least an attempt at, such a theory.

During the tests on the S.51 it was found that in the stick-fixed condition the machine was stable at between 25 and 35 knots, slightly unstable at 50 knots, and very nearly neutrally stable at 65 knots. On each occasion there was no need for the pilot to resume control within about 1½ minutes; so that it could not be said either that the stability or the instability was very marked.

Mr. Stewart had referred to the effect of the fuselage on the dynamic stability of the helicopter; the fuselage of the S.51 was much cleaner than the Hoverfly I, and he wondered whether they were getting nearer the behaviour of the rotor itself on the S.51.

At the A.F.E.E. they were also concerned with problems of the operational use of helicopters and were engaged in blind flying work. A Hoverfly I had been fitted with day/night flying equipment and some quite intensive flying had been done, which confirmed the opinion that the artificial horizon lost its primary importance and the air speed indicator became much more important. He would plead for that much-maligned instrument, the rate of climb indicator, which, on a helicopter, could take the place of the longitudinal indication of the artificial horizon—to a certain extent. If a neat, reliable and simple flight path indicator could be provided it would certainly provide the pilot with the information he required. But the production of such an instrument appeared to be unlikely for some years to come.

The Hoverfly I could not hover at altitude, but measurements had been made at between 30 and 70 m.p.h. forward speed in level flight, and it had been found that between 130 ft. and 300 ft. might be lost in building up the rotor r.p.m. in auto-rotation sufficiently to enable a safe landing to be made. So that, allowing for the actual landing, something of the order of 200 ft. might be the lowest height from which a safe landing could be made in the case of engine failure, even in forward flight.

O. L. L. FitzWilliams (Westland Aircraft Ltd.): In the early stages when the loss of control mentioned by Mr. Stewart and Lt.-Cdr. Brown was first experienced it was definitely frightening because the particular nature and layout of the Sikorsky R-4B gave rise to quite violent rolling and pitching moments. The reasons for those violent moments were not at first known, and there was some suspicion that they were caused by action of the air on the rotor blades of a kind which might have involved unusually rapid movements about the drag hinge and thereby have caused the hydraulic dampers acting about that hinge to impose high bending moments on the blade root. Mr. Stewart and Lt.-Cdr. Brown had shown considerable courage in repeatedly entering that highly unpleasant state.

He agreed with Mr. Stewart's remarks on stability; neither he nor most of the others concerned with the design of helicopters knew with any certainty how to achieve it, but he felt that it was going to be necessary to have positive stability of all kinds in all regions of flight. To confine attention to the use of a tailplane in forward flight as a solution of the stability problem was not really a solution even for blind flying, except as an immediate expedient, because they would soon be asked to provide helicopters suitable for steep

blind approaches down a beam, and a tail-plane would be of no use in that condition.

He agreed with Lt.-Cdr. Brown that the helicopter was already a reasonably easy machine to fly. Existing experience of training helicopter pilots indicated that, whether or not they had flown aeroplanes before, a pilot could fly solo after six to eight hours, and after perhaps 25 hours altogether was considered able to carry passengers. These figures were identical with those considered usual in pre-war flying clubs using Moths.

The chief trouble was not instability, but the unsymmetrical arrangement of the controls in the present aircraft, and it was difficult to overcome immediately. Every time the pilot wanted to go up or down he must move the pitch lever, which was connected to the throttle, so that he upset the torque balance and had to make compensating movements with all the other controls. While so far that was unavoidable, it was nevertheless a bad feature. He believed that this trouble would be eliminated and the helicopter made stable so that any person who was accustomed to being in the air should not need more than three hours of instruction at the outside in future before flying solo in a helicopter.

He did not consider that the problems of blind flying were very serious; he believed the existing instruments were quite adequate, especially if the helicopter were made stable, and agreed that the artificial horizon was not a primary instrument for helicopters, although it would be needed so that the pilot could relax. It was important that the pilot should be able to relax in steady flight, using the artificial horizon as a convenient reference, but the settling of the conditions of steady flight would have to be done by the other normal instruments. Perhaps the only qualification was that in the helicopter the present types of side-slip indicators could not be made to work satisfactorily by pendulum or bubble, and some other instrument would be necessary in the form of a yaw meter. Mr. Stewart had mentioned in the paper that he had such an instrument under development.

R. A. C. Brie (British European Airways, *Assoc. Fellow*): It had long been established that any given rotor system was susceptible to slight changes in parameter. Small refinements in detail design occasionally had the most surprising results, and the parallel theoretical and practical approach adopted by Mr. Stewart was the most logical and direct way of achieving those desirable improvements in stability and control characteristics which the helicopter so badly needed.

In discussing the effect of engine failure during take-off or landing, the lecturer referred to the possibility of some modification to existing piloting technique. What was the nature of this modification and had it been tried out in practice? There was evidence to support the view that as soon as rotor disc loadings were reduced to a more reasonable order, the problem such as it was would automatically solve itself. The existing obligatory method of powered landings called for far too much skill and precision and the time must come when, except for commercial operation in very restricted areas, it would be normal, rather than abnormal, for power-off landings to be made.

What was the definition, in terms of altitude, of "safety height"? The variables to be considered were those brought about by the natural wind velocity, the speed of the aircraft, its altitude, and whether it was nose-into or down wind. A helicopter which was down wind when the engine failed obviously required more air space to manœuvre than one already into wind. Had down-wind power-off tests been made to determine the loss in altitude on a 180° turn at various indicated air speeds, and if so what was its nature?

More precise information was needed also on the loss of control during vertical descent. Despite the many thousands of hours which had been flown there was no recorded instance of this phenomenon asserting itself during an approach to land. On the other hand, it was established that a nose pitching moment occurred at ultra slow speeds if the heading were down wind; the only method of recovery without appreciable loss in altitude being through an increase in power. He suggested that it was more appropriate to infer a loss of effective control, rather than complete loss of control.

It was generally acknowledged that the Sikorsky R-4 had peculiar aerodynamic and mechanical characteristics. The more recent S.51 model represented a marked improvement.

The ability to fly blind would appreciably enhance the utilisation value of the helicopter, but it was difficult to see how this would be satisfactorily accomplished unless some

simple and effective means was first devised to enable the pilot to compensate for off-set control column loads.

J. S. Shapiro (Cierva Autogiro Co. Ltd., *Assoc. Fellow*): The measurements taken in the hovering state were a striking confirmation of a simple theory which could be used by the designer and endowed this theory with sufficient authority in full-scale application to justify its use in design refinements. In his opinion, the experimental confirmation of tip losses theoretically predicted in this instance, proved the vortex theory all the more because the experimental figure was a difference between two large magnitudes. In such a case, agreement might be more difficult to obtain but if it were obtained, unless the test results showed large scatter from several tests, the confirmation was the more impressive.

In his treatment of vertical descent, Mr. Stewart had given some observations obtained under difficult and sometimes dangerous conditions, but had associated his description with theoretical generalisations. He thought these were somewhat at variance with hitherto accepted views. Were the deviations from older theories intentional and based on new undisclosed information, or were they an arguable interpretation of known facts?

The statement that simple momentum theory broke down as soon as a descent set in from the hovering condition, was a novel conception. Simple momentum theory was limited by a condition in which the air was brought to rest at infinity (relatively to the rotor). The momentum theory actually "broke down" before that, but the excellent agreement between measurement and momentum theory in the hovering state revealed by the lecturer would certainly suggest that momentum theory was entirely fulfilled in hovering and only began to "break down" at an appreciable rate of descent, always provided that momentum theory was applied in strip form and tip losses were taken into account.

Credit must be given to Dr. Bennett for having first applied the empirical relations developed by Glauert and Lock for the vortex ring and turbulent states in strip form to a rotor, and having thus pointed out that a descending rotor had different working states along the blade. The application of a strip method in the vortex ring state might sound paradoxical but might still be a useful working theory. The Cierva Autogiro Co. had developed numerical procedures for handling blades of any shape, using a quadratic substitute function for profile drag and applying the experimental results of Glauert without recourse to substitute functions. Their treatment was certainly better than the assumption of uniform induced velocity made by the lecturer in his "thrust method" and it allowed the taking into account of tip losses. Such methods at least provided an answer to those problems of vertical descent which were of primary interest to the designer, such as:

What was the parachutal efficiency of a particular rotor? Was blade twist, such as used for improvement of hovering efficiency, beneficial to vertical descent or otherwise? What was the limiting blade pitch for vertical auto-rotation? Did the rate of descent, anywhere in the time interval between loss of power and attainment of steady rate of vertical descent, exceed the steady rate?

Some of the above practical aspects could be directly checked experimentally, thus providing information on whether the empirical strip method was a useful tool in the hands of the designer.

Perhaps a more satisfactory approach would be the assumption of a certain distribution of vorticity giving rise to a flow pattern similar to that observed by means of smoke streamers but, in his opinion, flight testing in this matter should be principally concerned with the checking of conclusions resulting from existing or new theories; in other words, in this instance an inverse method should be applied in testing. In particular, he could not see how the curve by Mr. Hafner, referred to by the lecturer, could be applied to the hovering state in which the lecturer had found such excellent agreement with the vortex theory, while the Hafner curve departed considerably from the vortex theory in the hovering state. He suggested that this departure might have been derived from experiments and might be due to lack of consideration of vertical drag on the fuselage and inadequate estimates of tip losses.

Certain references made in the paper with regard to exorbitant energy losses in vertical descent seemed misleading. The total energy of the slipstream in every type of reactive propulsion was invariably lost whether the slipstream proceeded towards its destruction

in an orderly manner or broke up into macroscopic eddies. The relevant question was, what mass of air was associated with the downward momentum imparted by the rotor and in this respect vertical auto-rotation was a miraculously favourable condition, as compared with that of the simple momentum theory. The difference was best expressed by the parachutal coefficient which, in vertical descent, could rise up to two, whereas the limiting value of simple momentum theory was one.

Why was the vertical descent initiated from horizontal flight and not from hovering, since apparently hovering was a possible condition of flight when the machine was stripped of certain components?

He supported the request for dynamometric power measurement to be made. In his experience it was possible, by adequate estimation of tip losses and due consideration to the effect of Mach number on the lift slope of the blade profile, to predict hovering blade angle and r.p.m. with an accuracy of one per cent. But prediction of power and of lifting efficiency was not yet possible beyond an accuracy of four to six per cent. This was not good enough and should be remedied as soon as possible. Hovering did not constitute a large proportion of the flying life of a helicopter, but in a properly designed machine it determined the payload and this was now economically of vital importance.

The use of dynamometric power measurements was also essential for forward flight performance testing which could serve to establish reliable methods of analytical prediction. It was vital to know the parasite drag of the machine and, in his opinion, no performance analysis could be considered satisfactory which worked with a constant blade profile drag and thus almost entirely obscured the important effect of tip speed. Most helicopters had been disappointing, so far, in the fulfilment of performance predictions in forward flight and this matter was of equally vital importance economically. Much useful work could be done in this line without thorough investigations of the distribution of induced flow in forward flight and it seemed to him that a really useful approach to the latter problem must again proceed by an inverse method. H. B. Squire's paper to the Helicopter Association* had referred to an analysis which was laying the groundwork for such an inverse method. He believed that, to begin with, it was more profitable to observe flapping motions and blade deflections and to correlate them to simplified assumptions of induced flow distribution in forward flight, rather than begin by developing techniques for the measurement of bewildering patterns of flow.

He was surprised by the lecturer's figures on ground effect and would like to know more. Although he had no experimental evidence of his own, he had recently tried to survey available data from all sources and found that anything beyond 12 per cent. increase of lift at constant power at one half rotor diameter was controversial and that information from different sources varied considerably. Was the figure given in the paper for blade loading a misprint, as it was understood that present blade loading was around .20 and not .06?

It should be stressed that there was an element of instability in the ground effect. Even at walking pace, the slipstream which at zero speed spread circularly, suddenly rolled up and proceeded all in one azimuth direction. The aircraft, as it were, "rolled off" the ground cushion. Ground effect and wind did not, therefore, add up but a wind of the order of 10 m.p.h. was equivalent to ground effect and "took over." This rolling off the ground cushion was noticeable when the R-4 took off and began to fly forward.

He agreed with those who had criticised the reverence for stick-free stability. In his opinion such a notion was either meaningless or non-existent, according to whether the rotor control was reversible or irreversible. In the former case, stick forces were determined by accidental conditions, mainly errors in manufacture. An artificial stick-free stability could be obtained by the use of springs or any other devices.

It was interesting that c.g. position had been found of little effect on stick-fixed stability. He had contended for some time that in a single-rotor machine, the essential parameter with regard to the c.g. was its vertical distance from the rotor and not its horizontal position. C.g. limits on a single-rotor machine were determined by the danger of running out of control and not by the effect on stability.

Why had the name "phugoid motion"

* "Air Flow Through Rotors," H. B. Squire. Lecture given to the Helicopter Association on 6th December 1947.

been chosen to describe the evolutions of a helicopter with a fixed stick after a disturbance? To his knowledge, phugoids were solutions of the differential equation describing the motion of an aeroplane on the assumption that its lift was always at right angles to the flight path and equal to the square of its linear velocity, and neither of these conditions was fulfilled in the small disturbance motions of a helicopter superimposed on steady flight.

With the confidence acquired in analytical predictions of dynamic stability, the designer could develop improved stability. The three-rotor helicopters, at present in the course of construction by the Cierva Autogiro Co., were expected to be dynamically stable as well as to possess a stick-fixed static stability under all conditions of flight.

He was not quite clear what type of gust the lecturer had meant. If he referred to an up gust, its overall effect on the aircraft was equivalent to a tilt of the disc or change of disc incidence. Such a change produced an increase of lift but its effect on blade flapping and consequently on the physical tilt of the disc relative to the aircraft, was in any case small and depended on a variety of circumstances.

The lateral tilt could only be a secondary effect but, theoretically, the most noticeable difference between the effect of gusts on helicopters and fixed-wing aircraft was that the former had a much smaller lift slope and were therefore less sensitive. Another fact was the relative lack of sensitivity with respect to horizontal gusts.

Did not the artificial horizon remain a necessity even though it lost its secondary function of indicating the flight path? The pilot still wanted an indication of which way up he was and he could only have this information from a gyro horizon.

His own order of preference for flight test work in the near future would place accurate measurements of performance and further work on stability at the top of the list, followed by blade flapping and an evaluation of load factors.

He believed that detailed investigations into flow patterns and development of aerofoil sections could be postponed at this juncture if a choice had to be made.

Dr. J. A. J. Bennett (Fairey Aviation Co. Ltd., *Fellow*): The full-scale measurements to determine the relationship between the classic parameters f and F in the hovering state and in the adjoining vortex ring and propeller states were the first data on the subject since Glauert and Lock's analysis of propeller experimental data more than 20 years ago. The variables now adopted by Mr. Stewart were similar to those proposed by Lock in a recent note on the subject, namely:—

$(1/f)^{\frac{1}{2}} = V\,(2\rho/w)^{\frac{1}{2}}$ and $(1/F)^{\frac{1}{2}} = u\,(2\rho/w)^{\frac{1}{2}}$

where V was the axial velocity at ∞ and u the velocity through the disc. Whatever method of plotting was used, the critical information required was the values of those parameters for the hovering case. Using Glauert's empirical value of F for that case, where $1/f = 0$, the induced velocity v was related to the disc loading w and density ρ by the equation $v = \sqrt{w/\rho}$, which was approximately equal to the speed of descent in auto-rotation. The momentum theory, assuming a uniform distribution of induced velocity, gave the value $v = \sqrt{w/2\rho}$. In practice it was found that the Glauert value was about as pessimistic as the momentum theory was optimistic, and that the induced velocity was $v = \sqrt{w/k\rho}$ where $k \fallingdotseq 1.5$. That seemed to agree with the picture which Mr. Stewart had shown.

It had been found in the United States that the loss of control in the vortex ring state was not confined to vertical descent; it could occur at air speeds between 0 and 60 m.p.h. in the Sikorsky R-4 when the aircraft was deliberately held in the state of vibration peculiar to the vortex ring state. That state was usually avoided by pilots because of the associated vibration, but when the aircraft was held in that state for a few seconds it became nose heavy and there was a loss of control exactly as found by Mr. Stewart at zero air speed.

The vortex ring state occurred in a partial-power descent at critical sinking speeds decreasing from 600 ft. per minute at zero forward speed to zero at about 60 m.p.h., the amplitude of vibration being greater the lower the forward speed. The frequency was three times per revolution and the blades made a noise familiar to anyone who had watched the slow speed flight of rotary wing aircraft.

He was interested in Mr. Stewart's remarks about the top speed of the helicopter being limited by excessive flapping and vibration. Presumably he had meant by that the higher harmonics of flapping, *i.e* a simultaneous

upward displacement of all the blades. The vibration caused by periodic blade-tip stalling should be minimised in the gyrodyne due to the reduction in axial flow through the rotor. It would thus be of interest to evaluate the high speed operation of the gyrodyne configuration at a later stage of its flight trials.

J. Fearn (Westland Aircraft Ltd., *Assoc. Fellow*) *contributed*: He and his colleagues agreed with Mr. Stewart's remarks on the desirability of positive stability and believed this could be achieved without undue complication, although pressure of other work on the limited design staffs available might delay a satisfactory solution. In the meantime the thorough training of helicopter pilots fortunately made it possible for them to operate successfully without the advantage of positive stability. In cruising flight the stability characteristics of their S-51 seemed to be better than had been anticipated and it could be flown "hands off" in certain conditions for about 15 seconds without noticeably deviating from straight flight. In this case the deviation was caused by gradual change of stick position, since the controls were not quite irreversible. It was believed that stability trials so far conducted might be slightly misleading because of small movements of the controls which were sufficient to cause big changes in the motion of the aircraft, and these small motions were difficult to eliminate so long as the pilot's hand remained on the control stick.

They had hoped to have a more positive statement on the effect of adding a tailplane to a helicopter, such as the R-4b and would be interested to know Mr. Stewart's views on this question, especially in view of the reported favourable behaviour of the Bristol 171 to which a tailplane was fitted.

In one place Mr. Stewart referred to the "height of rotor above the ground," and in another place his reference appeared to be to the height of the wheels above the ground. Sikorsky used a phrase "hovering ceiling with ground effect" which referred to the altitude of the highest ground from which a helicopter could rise vertically in still air until its wheels were 10 ft. from the ground. This represented a useful practicable condition in which take-off would be unaffected by normal obstructions and it would be a good idea if some such convention were standardised. The beneficial effect of the ground cushion could only be discussed in relation to the height of the rotor above the ground and they agreed that this beneficial effect disappeared when the height was equal to one rotor diameter.

Was Mr. Stewart's reference to flying "under the hood" literally true? There must be a considerable difference between this and the two-stage amber scheme used in other experiments. They agreed that the artificial horizon did not seem to be a primary instrument for helicopter blind flying. They thought its main use would be to allow the pilot to relax in steady flight, steady conditions being attained by reference to other instruments. Their brief experience of the side-slip indicator on the Beaulieu aircraft was unfavourable as it appeared too sensitive for normal flying. They thought this instrument might be of considerable importance.

Mr. Stewart's remarks on the proper position for the pitot system were interesting. The position used by Sikorsky was satisfactory for contact flying but a more reliable system would be welcomed. Helicopter pilots were almost unanimous in requesting a sensitive low speed indicator and information on this point would be welcomed.

Had any other method, apart from the plumb line referred to by Mr. Stewart, been tried for indicating true vertical flight? This would be of great value to the helicopter pilot in those cases where he had a limited vertical view and could not look out of a window to observe what was happening to a plumb line.

W. E. Gray (*contributed*): He would like to urge an extension of the type of work described in this lecture—that was, where a close visual check was kept on the physical happenings during an experiment.

About a year ago he had urged Mr. Stewart to explore the flow through the rotor disc by means of smoke, and had helped him with the first tests, and he wished to congratulate him and his team on the thoroughness with which they had followed up the suggestion.

Too often in the past experiments in aerodynamics had got into troubled waters or run off the rails through too much theory and too little checking up on the physical happenings. The lecture was a happy blend of theory and verification.

Dr. G. S. Hislop (British European Airways, *Assoc. Fellow*) *contributed*: He had been associated with the work described by Mr. Stewart in its early stages and

complimented the lecturer on the successful completion of several of the most important phases of aerodynamic research on helicopters. He would like to ask the lecturer's views on the following:—

(i) The use of a torquemeter to determine the actual power supplied to the main rotor would seem to be an experimental tool of urgent necessity. It would help to determine the exact shape of the empirical curve in the vortex ring vertical descent case, as well as being invaluable in assessing the general efficiency of a given layout. Was there much prospect of a torquemeter becoming available shortly for use in this class of work?

(ii) It was interesting to note that the flight path recorder used was actually that employed many years ago on the relatively slow aircraft in existence at that time.

Another somewhat similar case of old ideas being revived which might be of interest was the fact that the use of a double venturi for the measurement of air velocities under the rotor disc was suggested to him by Pannell's use of a somewhat similar instrument in airship work. Perhaps there were other instruments which were developed for airship work which might be of interest and of use in future research programmes.

(iii) The great importance which the lecturer placed on the work on the longitudinal stability of helicopters was echoed by the operators, as much of the success of the operation of helicopters in all conditions of season and weather relied upon the ability to fly aircraft without undue strain. A marked improvement of the stability characteristics of helicopters was urgently needed.

In view of the comprehensive results already obtained in dynamic stability, he would suggest that it would be of great interest were the tests to be extended to include the effect of a tailplane fitted to the R-4. This would enable the effect of such aerodynamic surfaces to be appreciated and would also show whether any additional advantage or disadvantage would accrue in the vertical descent case.

(iv) The lecturer's view about the main deficiencies of helicopter blind flying equipment, namely lack of an instrument to give the pilot a reliable method of relating attitude and speed and the necessity for some better form of longitudinal indication, was echoed by B.E.A. experience in this field. It was hoped that the work which the Experimental Helicopter Unit at Yeovil would be doing shortly would help to throw some light on the best solution to this problem.

J. K. Williams (Air Registration Board, *Assoc. Fellow*) *contributed*: Mr. Stewart had concentrated on the investigation into the induced air flow through the rotor to help in solving the problem of stability and control. There were equally urgent problems, such as the gathering of sufficient flight test data for estabishing reliable design criteria.

An indication was needed of the maximum manœuvring load factors attainable in flight by the fitting of statistical accelerometers. It was comparatively easy on helicopters to put on full control movement inadvertently under the extreme conditions of maximum r.p.m. The maximum pilot's effort would not be required as for a fixed-wing aircraft at high speed. High accelerations could be introduced on helicopters in this way. With adequate statistical data it should be possible to stipulate maximum acceleration values for design purposes in terms of rotor diameter and disc loadings to ensure the minimum risk of an accident, such as blade failure in flight, but at the same time without introducing redundant blade and other structure weight.

The B.E.A. Helicopter Unit at Yeovil were doing useful work in this respect by fitting accelerometers to their Sikorsky and Bell types in the simulation of postal routes in the west of England, but the extreme manœuvres mentioned above would not be simulated deliberately, and they did not wish to experience high accelerations.

Little reliable information existed on the behaviour of helicopters in gusty conditions. Present experience indicated that helicopters rode gusts very well, *i.e.* the movement with a gust was small. At present they could only guess at the gust intensity to be catered for and the corresponding gust alleviating factors to be assumed for the rotor blades.

The greatest danger, it seemed to him, would come from sudden lateral gusts. The extreme danger that would arise if a helicopter flew around a very high building from the sheltered to the windward side, say, of a skyscraper, or even the new hangar at Bristol, could be imagined.

The types of helicopter that could be envisaged operating on scheduled inland routes during the next few years would all be single-engined, such as the Sikorsky S-51, the Bell 47 and the Bristol type 171.

Two contingencies could be visualised in connection with engine failure: —

(a) Failure under hovering conditions under zero wind velocity, *i.e.* with no possibility of a forward glide being initiated. For every helicopter a danger zone existed approximately between 10 ft. and 400 ft. vertically above the ground, in which zone it would be impossible for the blades to attain the auto-rotative, *i.e.* windmill state, before the machine hit the ground.

This danger zone would have to be determined for each helicopter as part of the flight testing programme of the prototype. The extent of the zone would then be scheduled in the flight manual.

(b) Engine failure under conditions of forward flight. It was envisaged that emergency landing sites would be spotted alongside the scheduled route for such a case. The technique to be adopted would be to reduce immediately the pitch of the blades to the auto-rotative condition and then glide in to land. At a height of about 10 ft. from the ground, the pitch of the blades was suddenly increased to a maximum, *i.e.* a pull out was initiated and the machine then descended to the ground at a constant vertical velocity of descent up to 12 ft. per second.

They believed that the technique of engine-off landings should be an integral part of the flight testing of each prototype helicopter and also should form a part of the training of helicopter pilots. If the technique of engine-off landings were mastered, then it would eliminate the carrying of long travel undercarriages to cater for excessive vertical velocities of descent.

MR. STEWART'S REPLY

The discussion could be divided mainly into the sections he had used originally, *i.e.* the nature of the flow through the rotor disc, the stability problems and blind flying.

The fitting of a tailplane to the helicopter was part of the programme at Farnborough, but although the tailplane might be a satisfactory solution at the higher speeds, it still did not solve the hovering flight case or the very low air speed case. At the same time, its influence was mainly in the stick-fixed condition. With a rotor with inherent forces which caused the helicopter to diverge, the tailplane would have to be very large in order to have any effect against that.

There were many methods in connection with the stability of helicopters which were only now being tried, such as the Bell stabiliser bar where, by the use of gyroscopic forces, there was a form of automatic control, and the phugoid motion with the Bell stabilising bar could be made quite stable. The Hiller system had come forward recently, and preliminary calculations had shown it to be even more stable than the Bell.

With the tailplane having its influence in the stick-fixed condition, and using the form of automatic control he had mentioned, he believed that the solution to the problem would be found fairly soon. There still remained the worst problem, what happened when the pilot let go the stick, and the solving of that problem might occupy a much longer time.

Mr. Hafner had said that the curve he had produced was largely a guess. He would not call it a guess, but if it were, then it was a very inspired guess, and correlation with the flight tests, even though they were slightly higher showed very good agreement.

Mr. Hafner had referred to the use of reflex sections for obtaining zero C_m (pitching moment), but he would draw attention to production difficulties. If one had a reflex trailing edge section which would give zero C_m and the various characteristics which one might want, that section depended entirely for its behaviour on the reflex nature of the trailing edge. The present rotor blades were essentially of a flimsy nature, and any deformation of that trailing edge, either in production or by aeroelastic distortion, could completely ruin the properties of that section. As a result, it was almost certain that they would see a reversion to the symmetrical section.

The loss of control experienced during vertical descent had received quite a lot of attention, and he would mention the brilliant efforts of Lt.-Cdr. Brown not only in finding it, but in investigating it fully from the top end, the bottom end, and all other aspects, to learn how one could get into that condition. It had since been experienced with various other types, such as the S-51 and the Bell 47, so that it was apparently fundamental to all helicopters.

That loss of control had not been experienced under normal circumstances, mainly because the majority of the flying up to the present had been on the R-4. The

method of landing the helicopter had been to fly it in at a reasonable speed, at about 30 to 40 m.p.h., on a very flat approach and decreasing the air speed to zero. It was quite easy to visualise the condition, with a high-powered helicopter which could climb vertically, where it could be brought over the landing point at a height of 200 to 300 ft. and would then begin to reduce height vertically; in that case they could get into the loss of control state. That was one of the factors which might modify landing technique.

There was a safety height for the helicopter for engine failure. The alternatives when an engine cut on the helicopter were: one, to get into complete auto-rotation, which involved considerable loss of height; it was at the moment of the order of 200 to 300 ft. The other alternative, which could be used if the machine were very low when the failure occurred, perhaps only 30 or 40 ft., was to make use of the kinetic energy of the rotor to arrest landing. At between 30 and 300 ft. there was not sufficient height to establish auto-rotation and there was not sufficient kinetic energy to enable the machine to be lowered by that means from that height. So that there was that region in which engine failure was dangerous. That height region, of from 30 to 300 ft. at zero air speed, decreased as the air speed was increased, so that at 30 to 40 m.p.h. auto-rotation could be used without such loss of height. Generally a safety height was quoted for zero air speed.

On comparing the auto-rotation descent with the parachute, what appeared at first sight to be a contradiction was found, that the helicopter was the better. Thinking about it, however, if the helicopter were treated as a disc in ideal auto-rotation, all the air approaching that disc would be brought to rest, and they had the pure disc drag or parachute drag. But in the practical case, while the majority of the sections might be reducing the air velocity to zero, there was air getting through either at the tips or at the centre, depending mainly on the twist and design of the blades. The result was that some sections were working better than the flat plate drag. So that on comparison the result was very much in favour of the helicopter.

He agreed that the helicopter was easy to fly, but only in certain aspects. The present helicopters, although easy to fly, were very tiring, requiring considerable concentration. While one could be taught to fly in five or six hours, to fly with safety, *i.e.* one did not break the machine, considerable concentration was still required on the part of the pilot. He did not think any pilot who was present would say that he would let the stick go and pick up something from the floor, take his handkerchief out of his pocket and then look around to find what was happening to the helicopter; he would have found out long before! Also, it was fundamental to bear in mind that, in teaching someone to fly a helicopter, he was being taught to take it off the ground and put it back on to the ground by raising and lowering it, and not by charging along a runway at about 60 to 100 m.p.h. before taking the machine off or putting it down. The concentration necessary on the part of the helicopter pilot was purely a matter of stability and control and he believed there would be rapid improvements in that respect.

While they were flight testing at Farnborough they were comparing the results with theory as much as possible, and already on the automatic lines there were many approaches to the problem of improving stability. They had already shown that the ordinary configurations were unstable, and little could be done about that.

Ordinary methods, such as the fitting of tailplanes, still could not do very much at low speeds, although they could improve matters to some extent at high speed. But it would appear from theory that the fundamental cure for the helicopter would be arrived at by finding some term to introduce into the stability equations, depending not on the angular velocity of the helicopter, but either on its actual displacement or on an acceleration term. The Bell system of stabilising bar and the Hiller system were already working on that assumption, and had produced very useful results.

Thus, while much remained to be done, they were on the right lines for solving the problem of stability in control and should see the solution of the problem very soon.

Dr. Hislop mentioned the use of instruments originally developed for low speed aircraft and airship work. It might seem strange to resurrect instruments of over twenty years ago but they were, in fact, dealing with the same order of flight conditions on the helicopter that was experienced in the early aircraft and airship work.

The vertical string had been their only method of identifying zero air speed conditions. A similar form to that used in U.S.A., *i.e.* an opposed pitot yawmeter, was tried but appeared sluggish compared with the string method. There was also the doubtful influence of the slipstream giving a position error to the opposed pitot.

The use of torquemeters would be a great asset in flight testing work, but reliable instruments of this nature were not available yet.

The use of V-g recorders to determine the general nature of the loadings that would be normally encountered was of great assistance to the designers. They had learned with great satisfaction that B.E.A. helicopters had been fitted with such instruments; the information from these would be of greater importance than the results that would be obtained during the specialised test flying at Farnborough.

In the flight technique for the vertical descents, the approaches were always made from forward flight by reducing the air speed because of the limitations of the helicopter. It was true that some work was done in hovering flight but this was only possible on the R-4B helicopter at low altitude, in extremely low air temperatures (only possible on several days of the year), with the loading restricted to a pilot (no parachute) and only five gallons of fuel.

In the nature of the flow conditions, it was true that the flow did not change instantaneously at the boundary conditions and it was therefore possible to extend a theoretical analysis in strip form to a certain degree into the vortex ring state from the propeller and windmill brake regions. It was to be noted that simple vortex theory or constant induced velocity broke down at the boundaries.

In the ground cushion work, the height of the rotor above the ground was the main parameter in determining the thrust increase. Any reference to the height of the wheels above the ground was only an indication of the hovering height of the helicopter in the flight tests. The curves given were for zero air speed and the effect of a wind was not additive. In simple terms the effect of ground effect plus the wind effect might be considered as about constant up to a wind of about 10 m.p.h. and for the higher velocities, it could be considered that the wind had "blown the ground cushion away."

In the blind flying tests, the reference to flying "under the hood" was quite literal. Fabric screens were used to enclose the right hand cockpit so that the pilot was flying under conditions of absolute zero visibility.

In stability work, the stick-free conditions referred to what happened if the pilot released the stick. Naturally this depended on the type of controls and for true irreversibility stick-free became synonymous with stick-fixed conditions. Nevertheless, on the S-51 which was supposed to be irreversible, there was a slow creep of the controls on release, which influenced the helicopter's behaviour. The height of the c.g. was very important in helicopter stability but once the design layout was completed it was difficult to vary the vertical height of the c.g. sufficiently for research investigations.

The effect of a tailplane was more complicated than would initially appear. Its main purpose was the damping of the oscillatory motion. This was only achieved with the stick-fixed or with completely irreversible controls. If the stick-free behaviour were divergent (and it must be remembered that present irreversible systems did allow control creep) the tailplane might not be of any real assistance under these conditions. The effect of the tailplane on the stick position to trim was a function of the position and the flow through the rotor disc. If the tailplane were in free air (extremely difficult to obtain in practice) the effect would be to improve the static stability. Because of the influence of the flow from the rotor the tailplane could cause a marked deterioration in static stability, *i.e.* the stick would move back as the air speed was increased, and the magnitude of these movements could be large for certain tailplane positions. They would require further information on the flow through the disc in forward flight before satisfactory positioning could be established without making flight tests with the tailplane in various positions.

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EQUILIBRIUM RUNNING OF THE SIMPLE JET-TURBINE ENGINE

by

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SUMMARY.

THIS paper discusses the elementary use of non-dimensional parameters for analysing the performance of a simple jet turbine engine. The components of the engine—intake, compressor, combustion chamber, turbine and final nozzle— are linked together by means of the component characteristics. The possible running points are such that the characteristics are satisfied simultaneously when the whole of the engine is in equilibrium.

The components of the engine in general can be matched only at one point, called the design point. Performance at equilibrium points away from the design point may be predicted by the methods described, before the engine is built.

The paper also shows how the essential parameters governing the behaviour of the jet turbine engine are linked with the aircraft parameters.

1. INTRODUCTION.

DIMENSIONLESS PARAMETERS.

Some important quantities well known in the theory of propulsive systems, for example Mach number M, specific heat C_p and compressor efficiency η, are true dimensionless parameters, *i.e.* they have the same value independent of the units of mass, length and time used.

A number of other dimensionless parameters are met with in studying the turbo jet. These parameters connect the principal quantities governing the flow of the gas through the intake, compressor, combustion chamber, turbine and final propelling nozzle of the system, and they may be built up by considering the dimensions of the quantities involved as shown in Table I.

We may introduce immediately three valuable numbers which have the form of a Mach number. These numbers are truly non-dimensional, as may be checked from the above table.

(a) $v/\sqrt{T}$ the "non-dimensional" gas velocity.

(b) $ND/\sqrt{T}$ the "non-dimensional" peripheral speed.

(c) $(Q\sqrt{T})/(D^2P)$ the "non-dimensional" mass flow.

As stated (a), (b) and (c) are essentially of the form of a Mach number. For since

$$M = \frac{\text{the local gas velocity}}{\sqrt{\gamma g R T}}$$

where $\gamma = C_p/C_v$ is the ratio of the two specific heats of the gas, g the gravitational constant,

TABLE I.

Quantity	*Symbol*	*Dimensions* [L] = Length [M] = Mass [t] = Time
Diameter of flow path	D	L
Gas pressure	P	M/t^2L
Gas temperature	T	L^2/t^2
Mass flow rate	Q	M/t
Gas velocity	v	L/t
Propulsive thrust	F	ML/t^2
Energy consumption rate (= fuel rate × heating value)	$(Q_F H)$	ML^2/t^3
Rotational speed	N	$1/t$

R the gas constant; we see that $V/\sqrt{T}$ and $ND/\sqrt{T}$ are proportional to the M at the appropriate point or periphery. Further since

$Q \propto$ (flow velocity $\times D^2 \times$ density)

$$\propto \frac{VD^2P}{T}$$

Then $(Q\sqrt{T}/D^2P) \propto (VD^2P/T)(\sqrt{T}/D^2P) \propto V/\sqrt{T}$

which is again of the form of a Mach number.

It is also possible to frame non-dimensional groups containing the thrust and fuel consumption. The most convenient are:—

$F/(D^2P)$ called the "non-dimensional" thrust

$(HQ_F)/(D^2P\sqrt{T})$ the "non-dimensional" fuel consumption.

We shall see later that the above two parameters are the natural basis for the expression of thrust and fuel consumption of a turbo-jet engine.

In dealing with any particular engine we are usually justified in leaving out the linear dimension D as this is always common. We also leave out the heating value of the fuel as this is also a common factor in the great majority of cases. Then somewhat condensed parameters may be formed, namely:—

$(Q\sqrt{T})/P$ the "non-dimensional" mass flow

$N/\sqrt{T}$ the "non-dimensional" peripheral speed

F/P "non-dimensional" thrust

and $Q_F/P\sqrt{T}$ "non-dimensional" fuel consumption.

Having introduced our quantities, we proceed to show their application to the theory of the simple jet.

2. "NON-DIMENSIONAL" PERFORMANCE EQUATIONS FOR THE SIMPLE JET.

The figure represents the simple jet turbine where the reference planes between the consecutive components are indicated by α, 1, 2, 3, 4 and 5. Here

(α) – (1) is the intake

(1) – (2) the compressor

(2) – (3) the combustion chamber

(3) – (4) the turbine

(4) – (5) the propulsive jet

We suppose it is in equilibrium, *i.e.* running at a steady speed.

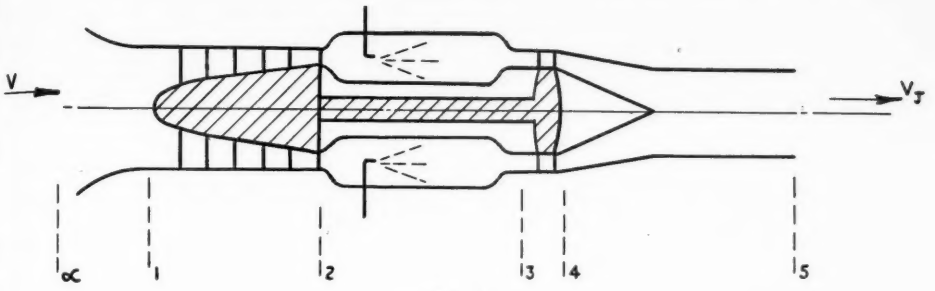

Fig. 1.
Reference planes.

2.1. THE INTAKE EQUATIONS.

Consider first the intake (∝) – (1). If V is the flight speed, then the K.E. available for compression in the intake may be expressed as a temperature rise, *i.e.*

$$T_{1t} - T_\alpha = V^2/(2gJC_p) \quad . \quad . \quad . \quad . \quad . \quad . \quad (1)$$

Here T_{1t} is the total head temperature of the stream at any point up to and including reference plane (1).

To calculate the pressure rise obtained from the intake ram we assume $\eta_{\alpha 1}$ is the isentropic efficiency and then the temperature rise available for isentropic compression is

$$T_{1t'} - T_\alpha = (\eta_{\alpha 1})\, V^2/(2gJC_{p_{\alpha 1}}) \quad . \quad . \quad . \quad . \quad (2)$$

where T_{1t}' is the hypothetical temperature at plane (1) for isentropic compression in the intake which defines the pressure ratio P_{1t}/P_α thus

$$P_{1t}/P_\alpha = (T_{1t}/T)^{\gamma/(\gamma-1)} \quad . \quad . \quad . \quad . \quad . \quad . \quad (3)$$

Now from equation (2)

$$T_{1t'}/T_\alpha = 1 + \eta_{\alpha 1} \{ V^2/(2gJC_{p_{\alpha 1}} T_\alpha) \} \quad . \quad . \quad . \quad . \quad (4)$$

$$\text{Whence } P_{1t}/P_\alpha = \{ 1 + \eta_{\alpha 1} V^2/(2gJC_{p\alpha 1} T_\alpha) \}^{\gamma/(\gamma-1)} \quad . \quad . \quad . \quad (5)$$

Equations (4) and (5) give the ram pressure and temperature ratios in terms of $\eta_{\alpha 1}$ and $V/\sqrt{T_\alpha}$. Both equations are truly "non-dimensional." The parameter $V/\sqrt{T_\alpha}$, which is proportional to the flight Mach number may be regarded as the "non-dimensional" forward speed.

Note that the full intake temperature rise $T_{1t} - T_\alpha$ or $V^2/2gJC_{p_{\alpha 1}}$ will be obtained whatever the isentropic efficiency.

T_{1t}/T_α and P_{1t}/P_α may be plotted against $V/\sqrt{T_\alpha}$ as in Fig. 2. These curves constitute the characteristics of the intake.

We may calculate $V/\sqrt{T_\alpha}$ from given flight conditions, then (if we know $\eta_{\alpha 1}$) we can read off P_{1t}/P_α and T_{1t}/T_α. A flight condition in this sense means a forward speed V and the atmospheric pressure P_α and temperature T_α.

2.2. THE COMPRESSOR.

Compressor performance characteristics give the pressure ratio P_{2t}/P_{1t} and temperature rise ratio $(T_{2t} - T_{1t})/T_{1t}$ in terms of $N/\sqrt{T_{1t}}$ and $Q\sqrt{T_{1t}}/P_{1t}$. As already explained these two numbers are of the form of Mach numbers governing the peripheral speeds and through flow of the compressor. The compressor characteristics will be known in any particular case. They will be of the form shown in Fig. 3.

Here the separature curves are for constant values of $N/\sqrt{T_{1t}}$. They are connected through the peak points by the surge line. Stable performance is possible only if the operating point is on the right of the surge line.

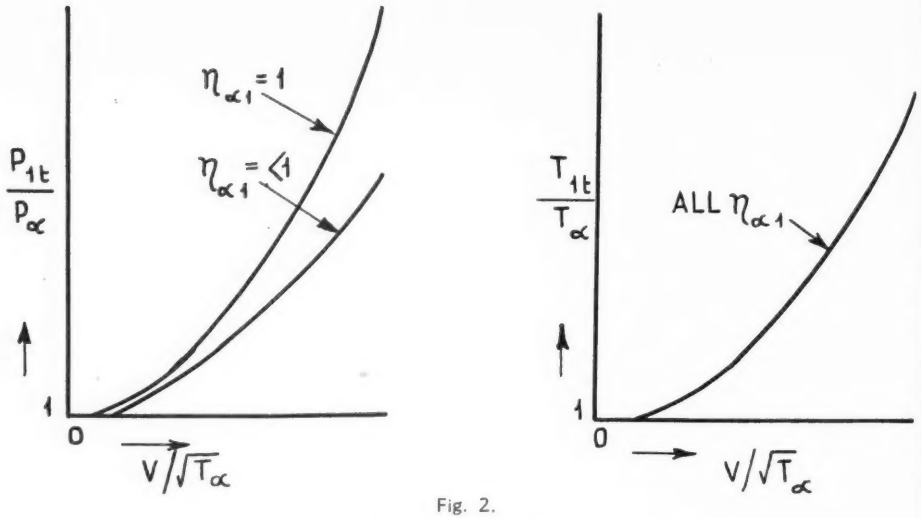

Fig. 2.
Intake characteristics.

We assume, at this stage that we know the mass flow Q. We may then calculate $Q\sqrt{T_{1t}}/P_{1t}$ since

$$(Q\sqrt{T_{1t}}/P_{1t}) = (Q\sqrt{T_\alpha}/P_\alpha)\,(\sqrt{T_{1t}}/T_\alpha)\,(P_\alpha/P_{1t}) \quad . \quad . \quad . \quad (6)$$

and T_α, P_α, $\sqrt{(T_{1t}/T_\alpha)}$ and P_α/P_{1t} are known.

We may also calculate $N/\sqrt{T_{1t}}$ from N, T_α and T_{1t}/T_α.

With $Q\sqrt{T_{1t}}/P_{1t}$ and $N/\sqrt{T_{1t}}$ we can plot the chosen running point on the compressor diagram. It is a chosen point in the sense that we have had to assume the value of Q in order to start on the problem. Having located the running point we can read off

$$(T_{2t} - T_{1t}/T_{1t}) \text{ and } (P_{2t}/P_{1t}).$$

From these quantities we may obtain $Q\sqrt{T_{2t}}/P_{2t}$ thus

$$(Q\sqrt{T_{2t}}/P_{2t}) = (Q\sqrt{T_{1t}}/P_{1t}) \times (P_{1t}/P_{2t}) \times (\sqrt{T_{2t}/T_{1t}}) \quad . \quad . \quad . \quad (7)$$

This number $Q\sqrt{T_{2t}}/P_{2t}$ represents the mass flow at the inlet to the combustion chamber.

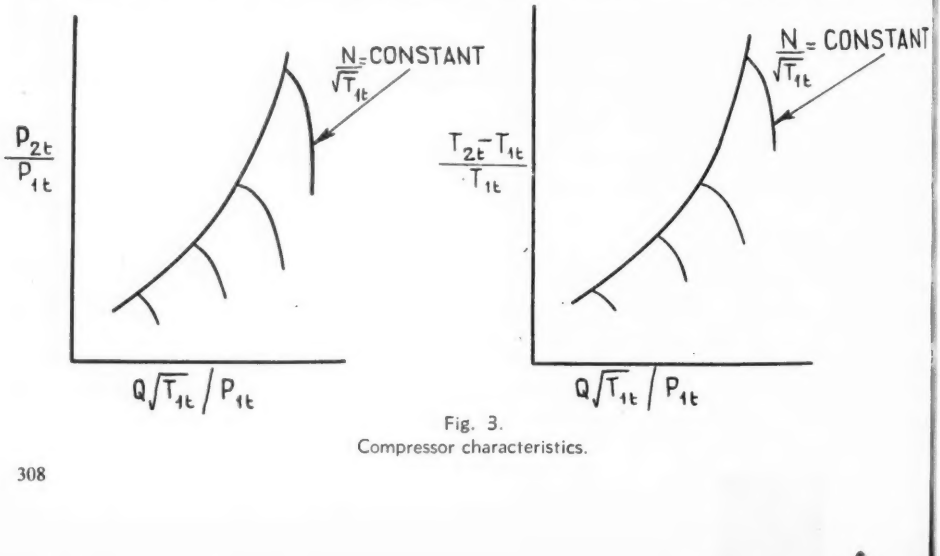

Fig. 3.
Compressor characteristics.

2.3. THE COMBUSTION CHAMBER.

The outlet temperature from the combustion chamber T_{3t} is usually fixed because of material limitations. T_{3t} is then the known constant; but it is more convenient to regard T_{3t}/T_{1t} as the quantity expressing the total heat input to the air; since this parameter, we shall find, links the turbine characteristics with the compressor characteristics.

Usually some of the compressor delivery air is bled off before the combustion chamber is reached to cool the turbine disc and bearings. A normal amount for cooling two per cent. of Q. Now this is of the same order as the fuel added; so that to a close approximation just as much gas leaves the combustion chamber as passes through the compressor. See Fig. 4.

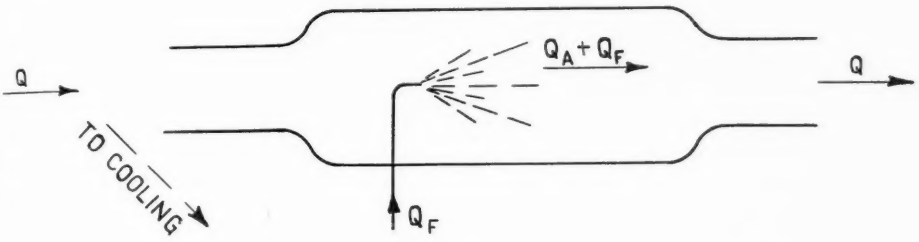

Fig. 4.
Combustion chamber.

If Q_A is the air mass flow rate in the chamber Q_F is the fuel mass flow rate.
Then with the above assumption we may write

$$Q = Q_F + Q_A \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (8)$$

The heat added per second is $Q_F H$ and this raises the temperature of the air and fuel vapour from T_{2t} to T_{3t}. It also has to vaporise the fuel but generally the latent heat is small enough to be neglected.

Let η_c be the combustion efficiency, *i.e.* the fraction of the heat energy of the fuel which is released in the combustion chamber; then from the heat balance we have

$$\eta_c Q_F H = (Q_A + Q_F) C_{P2\cdot3}(T_{3t} - T_{2t}) \quad . \quad . \quad . \quad . \quad . \quad (9)$$

where $C_{P2\cdot3}$ is the mean specific heat at constant pressure for the air and fuel vapour mixture over the heating range $T_{3t} - T_{2t}$.

Let the mean fuel/air ratio in the combustion chamber be q then

$$q = \frac{Q_F}{Q_A} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (10)$$

and

$$\eta_c q H = (1 + q) C_{P2\cdot3}(T_{3t} - T_{2t})$$

whence

$$q = 1/[(\eta_c H)/\{C_{P2\cdot3}(T_{3t} - T_{2t})\} - 1] \quad . \quad . \quad . \quad . \quad (11)$$

Since $C_{P2\cdot3}$ is a known function of the actual temperatures T_{3t} and T_{2t} we may construct curves giving q for any heating range $(T_{3t} - T_{2t})$ for various values of T_{3t} and η_c. Alternatively, with negligible loss of accuracy we may modify equation 11 to write

$$\eta_c q = 1/[H/\{C_{P2\cdot3}(T_{3t} - T_{2t})\} - 1] \quad . \quad . \quad . \quad . \quad (12)$$

and then we need only one set of curves giving $\eta_c q$ against $T_{3t} - T_{2t}$ for various T_{2t}. Having read off $\eta_c q$ and knowing η_c we have q (Fig. 5).

To estimate the outlet pressure P_3 of the combustion chamber we allow for a pressure drop, $P_{2t} - P_{3t}$ from the compressor delivery; and since this loss is primarily due to turbulence and friction of an aerodynamic nature, we may write

$$P_{2t} - P_{3t} \propto \rho_2 V_2^2 \quad . \quad . \quad . \quad . \quad . \quad . \quad (13)$$

$\frac{1}{2}\rho_2 V_2^2$ being the dynamic head at entry to the combustion chamber.

Now $\rho_2 V_2^2$ is the same as $\gamma P_2 M_2^2$ and M_2 is proportional to $Q\sqrt{T_2}/P_2$.

Equation (13) may therefore be written

$$P_{2t} - P_{3t} \propto P_2 \{ (Q\sqrt{T_2})/(P_2) \}^2$$

or
$$P_{3t}/P_{2t} = 1 - \text{a constant} \{ (Q\sqrt{T_2})/(P_2) \}^2 \quad . \quad . \quad . \quad . \quad (14)$$

This law is only approximate and for our convenience we rewrite it as

$$P_{3t}/P_{2t} = 1 - \text{a constant} \{ (Q\sqrt{T_{2t}})/(P_{2t}) \}^2 \quad . \quad . \quad . \quad . \quad (15)$$

neglecting the difference between the static and total pressures and temperatures at the entrance to the combustion chamber where the Mach number is invariably small. The constant will in any case be determined from known data and thus (15) will be strictly true in the region where this applies.

Having found $Q\sqrt{T_{2t}}/P_{2t}$ from equation 7, (15) may be used to find P_{3t}/P_{2t}.

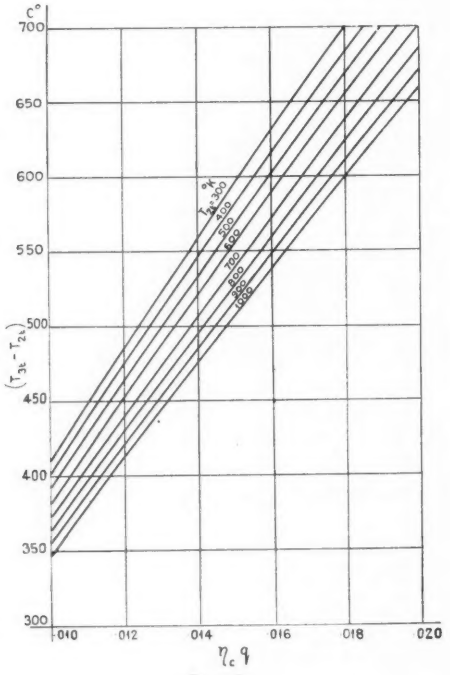

Fig. 5.
Estimation of fuel/air ratio.

2.4. THE TURBINE.

The turbine characteristics will give P_{3t}/P_{4t} and $(T_{3t} - T_{4t})/T_{3t}$ as functions of $Q\sqrt{T_{3t}}/P_{3t}$ and $N/\sqrt{T_{3t}}$. They will be something of the form given in Fig. 6.

Let us suppose we know the total heat input parameter—T_{3t}/T_{1t}. We may then calculate $Q\sqrt{T_{3t}}/P_{3t}$ and $N/\sqrt{T_{3t}}$ from data already established for the chosen operating condition. Thus

$$(Q\sqrt{T_{3t}}/P_{3t}) = (Q\sqrt{T_{1t}}/P_{1t})(P_{2t}/P_{1t})(P_{3t}/P_{2t})(\sqrt{T_{3t}/T_{1t}}) \quad . \quad . \quad (16)$$

$$(N/\sqrt{T_{3t}}) = (N/\sqrt{T_{1t}})(\sqrt{T_{1t}/T_{3t}}) \quad . \quad . \quad . \quad . \quad . \quad (17)$$

The results (16) and (17) are sufficient to locate the chosen running point on the turbine characteristics, and P_{3t}/P_{4t} and $(T_{3t} - T_{4t})/T_{3t}$ may be read off.

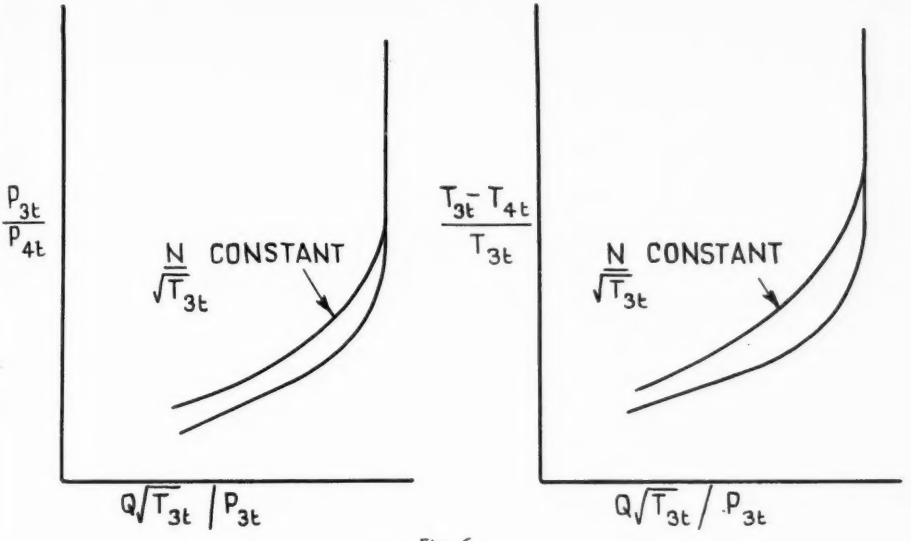

Fig. 6.
Turbine characteristics.

It is here that we have a check on the suitability of the chosen point. First, we have a value for P_{4t}/P_o since

$$(P_{4t}/P_o)=(P_{1t}/P_o)\,(P_{2t}/P_{1t})\,(P_{3t}/P_{2t})(P_{4t}/P_{3t}) \quad . \quad . \quad . \quad . \quad (18)$$

and to be suitable this must be greater than unity, otherwise it would be impossible for the gas to flow out of the propelling nozzle. Secondly, the balance of work between the compressor and turbine requires that

$$C_{P1.2}(T_{2t}-T_{1t})=\eta_m C_{P3.4}(T_{3t}-T_{4t}) \quad . \quad . \quad . \quad . \quad . \quad (19)$$

where η_m, the mechanical efficiency of the drive between the two units, is nominally about 99 per cent.

Equation (19) may be re-written in a more convenient form thus

$$(T_{3t}-T_{4t})/T_{3t}=(C_{P1.2}/C_{P3.4})\,(1/\eta_m)\,\{\,(T_{2t}-T_{1t})/T_{1t}\,\}\,(T_{1t}/T_{3t}) \quad . \quad . \quad (19)$$

Now for the chosen operating point to be possible, the value of $(T_{3t}-T_{4t})/T_{3t}$ calculated from the last equation must agree with that read off the turbine characteristics. If it does not agree we try another value of $Q\sqrt{T_{1t}}/P_{1t}$ keeping on the same $N/\sqrt{T_{1t}}$ line for the time being. When equation (19) is satisfied we have located the equilibrium point for the particular value of the parameter T_{3t}/T_{1t}.

This point can be plotted on the compressor and turbine characteristics since all the necessary pressure and temperature ratios are known. From the turbine characteristics we can now calculate the proper value of $Q\sqrt{T_{4t}}/P_{4t}$ since

$$(Q\sqrt{T_{4t}}/P_{4t})=(Q\sqrt{T_{3t}}/P_{3t})\,(P_{3t}/P_{4t})\,(\sqrt{T_{4t}/T_{3t}}) \quad . \quad . \quad . \quad . \quad (20)$$

The new quantity is the "non-dimensional" mass flow at the entry to the final propelling nozzle.

2.5. The propelling nozzle and tail jet.

The isentropic temperature ratio available for the final expansion in the propulsive jet, with no losses, is given by

$$(T_{F'}/T_{4t})=(P_o/P_{4t})^{(\gamma-1)/\gamma} \quad . \quad . \quad . \quad . \quad . \quad (21)$$

where $T_{F'}$ is the hypothetical final temperature after the expansion is taken right down to the static atmospheric pressure P_a. The isentropic heat drop obtainable with the above temperature ratio is

$$T_{4t} - T_{F'} = T_{4t} \{ 1 - (P_a/P_{4t})^{(\gamma-1)/\gamma} \} \quad . \quad . \quad . \quad . \quad (22)$$

This heat drop is not all available because of frictional effects which "re-heat" the gas to temperatures above the isentropic as it flows. The available heat drop is calculated from the isentropic heat drop by multiplying by the "isentropic efficiency" for the expansion process η_{4a}. Then

$$T_{4t} - T_F = \eta_{4a} (T_{4t} - T_{F'}) \quad . \quad . \quad . \quad . \quad . \quad (23)$$

This expansion may all take place inside the tail pipe, in which case the discharge from the final nozzle will be subsonic and P_5 equal to P_a; or it may take place partly inside and partly outside the tail pipe in which case conditions at the exit will correspond to the critical as limited by the speed of sound. P_5 will then be greater than P_a and will be so set as to yield the maximum rate of discharge.

The jet velocity derived from the heat drop of equation (23) follows from its K.E. equivalent namely:

$$V_j^2/2gJC_{P4a} = (T_{4t} - T_F) \quad . \quad . \quad . \quad . \quad . \quad (24)$$

V_j is the actual jet velocity since T_{4t} already allows for any dynamic head at the plane 4 where the jet enters the nozzle, and T_F is the actual static temperature calculated from the final static pressure after allowing for any energy losses.

Equations (22), (23) and (24) give

$$V_j^2/\sqrt{T_{4t}} = 2gJC_{P4a}\eta_{4a} \{ 1 - (P_a/P_{4t})^{(\gamma-1)/\gamma} \} \quad . \quad . \quad . \quad (25)$$

In any particular case we know P_{4t}, T_{4t} and P_a, so that given a value for η_{4a} we may calculate V_j.

The propulsive thrust F then follows from the change of momentum of the air stream in passing through the engine

$$F = Q(V_j - V)/g \quad . \quad . \quad . \quad . \quad . \quad . \quad (26)$$

From equations (8) and (10) the fuel consumption is given by

$$Q_F = Qq/(1+q) \quad . \quad . \quad . \quad . \quad . \quad . \quad (27)$$

and hence the specific fuel consumption is

$$\text{s.f.c.} = \{ q/(1+q) \} \{ g/(V_j - V) \} \quad . \quad . \quad . \quad . \quad (28)$$

Regarding units of the final quantities thrust and fuel consumption; Q is usually in lb. per second and the velocities in feet per second; q is non-dimensional. The thrust F is then in lb. and specific fuel consumption in lb. per lb. of thrust per second. The latter quantity is so small that it is usual to express s.f.c. in lb. per lb. of thrust per hour by multiplying equation (28) by 3600.

3. JET PIPE CHARACTERISTICS.

In the last section the final jet velocity was calculated from the full available pressure ratio P_{4t}/P_a. If the pressure ratio P_{4t}/P_a exceeds the critical corresponding to the final nozzle choking flow then P_5 becomes greater than P_a, and there is a discontinuity of pressure across the exit section. In this case the exit velocity V_5 is the result of the heat drop from P_{4t} to P_5 and the thrust is given by

$$F = QV_5/g + A_5(P_5 - P_a) - QV/g \quad . \quad . \quad . \quad . \quad (29)$$

where the first term is the momentum discharge, the second the force corresponding to the pressure differential at the discontinuity $(P_5 - P_a)$ and the third the intake momentum drag.

The expansion from P_5 to P_a which takes place outside the nozzle increases the jet speed from V_5 to the full V_j. The latter we have already calculated from the pressure drop P_{4t} to P_a by means of equation (25). The two expressions for the thrust given by equations (26) and (29), *i.e.*

$Q(V_j - V)/g$ and $Q(V_5 - V)/g + A_5(P_5 - P_a)$ must be equivalent,

although there will be slight differences in the value of the expansion efficiency for flow between planes (4) and (α) and between planes (4) and (5).

Following equation (25) the velocity at the exit section will be given by

$$(V_5/\sqrt{T_{4t}})^2 = 2gJC_{P4.5}\eta_{4.5}\{1-(P_5/P_{4t})^{(\gamma-1)/\gamma}\} \quad . \quad . \quad . \quad (30)$$

The mass flow is given by

$$\begin{aligned} Q &= A_5\rho_5 V_5 \\ &= (A_5P_5V_5)/(RT_5) \end{aligned}$$

We put the last equation into "non-dimensional" form thus;

$$(Q\sqrt{T_{4t}})/(A_5P_{4t}) = (1/R)(P_5/P_{4t})(V_5/\sqrt{T_{4t}})(T_{4t}/T_5) \quad . \quad . \quad (31)$$

Now $T_{4t}-T_5=\eta_{4.5}(T_{4t}-T_{5'})$

or

$$\begin{aligned} (T_{4t}-T_5)/T_{4t} &= \eta_{4.5}\{1-(T_{5'}/T_{4t})\} \\ &= \eta_{4.5}\{1-(P_5/P_{4t})^{(\gamma-1)/\gamma}\} \end{aligned}$$

$$\therefore \quad T_{4t}/T_5 = 1/[1-\eta_{4.5}\{1-(P_5/P_{4t})^{(\gamma-1)/\gamma}\}] \quad . \quad . \quad . \quad (32)$$

Substituting in the mass flow equation (31), the values of $V_5/\sqrt{T_{4t}}$ from equation (30) and T_{4t}/T_5 from equation (32), we have

$$\frac{Q\sqrt{T_{4t}}}{A_5/P_{4t}} = \frac{\{\sqrt{(2gJC_{P4.5}\eta_{4.5})/R}\}\{1-(P_5/P_{4t})^{(\gamma-1)/\gamma}\}^{1/2}(P_5/P_{4t})}{1-\eta_{4.5}\{1-(P_5/P_{4t})^{(\gamma-1)/\gamma}\}} \quad . \quad . \quad (33)$$

The last equation gives $(Q\sqrt{T_{4t}})/(A_5P_{4t})$ as a function of A_5, $\eta_{4.5}$ and P_5/P_{4t}. From it we may plot $Q\sqrt{T_{4t}}/P_{4t}$ for any given values of A_5 and $\eta_{4.5}$ as a single valued function of P_5/P_{4t}. (See Fig. 7.) $Q\sqrt{T_{4t}}/P_{4t}$ increases to a maximum as P_{4t}/P_5 increases up to which point $P_5=P_a$. According to the equation a further increase in the ratio P_{4t}/P_5 causes $Q\sqrt{T_{4t}}/P_{4t}$ to decrease. In actual practice the flow would remain at the choking value, the pressure P_5 rising automatically above P_a.

If there were no losses in the flow through the jet pipe it would choke when the exit velocity equalled the local velocity of sound (a_5). The heat drop in the nozzle would then be

$$T_{4t}-T_{5'} = a_5^2/(2gJC_{P4.5}) \quad . \quad . \quad . \quad . \quad . \quad (34)$$

Now

$$a_5^2 = \gamma gRT_5 \quad . \quad . \quad . \quad . \quad . \quad . \quad (35)$$

Therefore for ideal critical conditions at the jet pipe exit

$$T_{4t}-T_{5'} = (\gamma gRT_5)/(2gJC_{P4.5}) \quad . \quad . \quad . \quad . \quad (36)$$

Now

$$R/JC_P = (\gamma-1)/\gamma \quad . \quad . \quad . \quad . \quad . \quad (37)$$

and substituting into equation (36) gives

$$\left.\begin{aligned} (T_{4t}/T_{5'})_{crit} &= (\gamma+1)/2 \\ \therefore \quad (P_{4t}/P_{5'})_{crit} &= \{(\gamma+1)/2\}^{\gamma/(\gamma-1)} \end{aligned}\right\} \quad . \quad . \quad . \quad . \quad . \quad (38)$$

Because $\eta_{4.5}$ is less than unity, (T_{4t}/T_5) at the choking condition is slightly less than that given by equation (38), *i.e.* according to equation (33) sonic conditions are not quite attained. This, of course, conflicts with actual experience; but for efficiencies near to 100 per cent. the discrepancies are negligible.

The maximum "non-dimensional" mass flow for critical conditions may be found by substituting the results given in equation (38) into equation (31) whence

$$\{(Q\sqrt{T_{4t}})/(A_5P_{4t})\}_{crit} = \sqrt{(\gamma g/R)}\{2/(\gamma+1)\}^{\{(\gamma+1)/2(\gamma-1)\}} \quad . \quad . \quad (39)$$

On account of friction the mass flow at the choking condition is slightly less than this.

Going back to equation (33), the curve of the relationship between $Q\sqrt{T_{4t}}/A_5P_{4t}$ and P_{4t}/P_5 may be regarded as the characteristic of the jet pipe. Now it is more convenient to use P_{4t}/P_{1t} as the variable pressure ratio instead of P_{4t}/P_5 for then the characteristics are in a form suitable for linking with the compressor characteristics.

We have seen that the plot of $Q\sqrt{T_{4t}}/P_{4t}$ against P_{4t}/P_5 for a given A_5 and $\eta_{4.5}$ is a curve rising to a maximum mass flow and thereafter remaining horizontal. We may equally well regard this as a curve of $Q\sqrt{T_{4t}}/P_{4t}$ against P_{4t}/P_a; because below the choking value

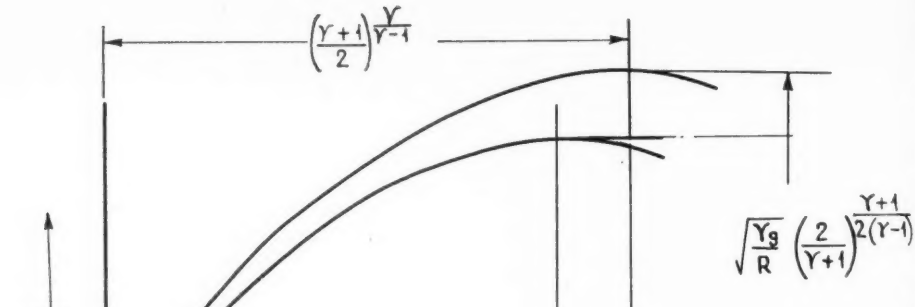

Fig. 7.
Plot of jet pipe flow equation.

P_5 and P_∞ are identical and above it there is only one value of $Q\sqrt{T_{4t}}/P_{4t}$ in any case. Hence if we plot $Q\sqrt{T_{4t}}/P_{4t}$ against P_{4t}/P_∞ we get a curve identical to that for $Q\sqrt{T_{4t}}/P_{4t}$ against P_{4t}/P_5.

This curve can in turn be converted to a base of P_{4t}/P_{1t}. Thus by means of the intake characteristics (Fig. 2), we have the relationship between P_{1t}/P_∞ and $V/\sqrt{T_\infty}$ for various values of intake efficiency. With a given intake efficiency and $V/\sqrt{T_\infty}$ we may read off $P_{1t}/P_\infty = y$, say.

Then

$$P_{4t}/P_{1t} = (P_{4t}/P_\infty)(1/y) \quad . \quad . \quad . \quad . \quad . \quad . \quad (40)$$

To convert to a base of P_{4t}/P_{1t} therefore, take the abscissæ of several points on the $Q\sqrt{T_{4t}}/P_{4t} - P_{4t}/P_5$ curve (Fig. 7), divide each by y and plot against the same value of $Q\sqrt{T_{4t}}/P_{4t}$.

When $V/\sqrt{T_\infty} = 0$, $y = 1$ and the new curve coincides with the old. At higher values of $V/\sqrt{T_\infty}$ the new curve comes to the left of the old an increasing amount as $V/\sqrt{T_\infty}$ is increased. Now the jet pipe has the same choking mass flow independent of the value of $V/\sqrt{T_\infty}$ and the various curves of constant $V/\sqrt{T_\infty}$ are tangential to the same choking value in Fig. 8. The choking points on the individual $V/\sqrt{T_\infty}$ lines; 0, 1, 2 and so on, correspond to the division of P_{4t}/P_5 by y_0, y_1, y_2 and so on.

The final curves, Fig. 8, are the jet pipe characteristics. These curves may be transferred to the compressor characteristics provided the engine is in equilibrium, as can be seen from the following.

In section 2, it was shown that the equilibrium running conditions determine the possible operating points on the compressor characteristics. The data used to fix an equilibrium point were the flight condition *i.e.* V, P_∞, T_∞ and N and the ratio T_{3t}/T_{1t}.

Having worked our way through the components we saw the way to calculate $Q\sqrt{T_{4t}}/P_{4t}$ and P_{4t}/P_∞, from the outlet conditions of the turbine, which parameters govern the jet pipe flow.

The jet pipe characteristics, Fig. 8, tell us the value of P_{4t}/P_{1t} required to sustain any definite mass flow $Q\sqrt{T_{4t}}/P_{4t}$ at any particular $V/\sqrt{T_\infty}$. Now we can calculate the value of P_{4t}/P_{1t} for all our equilibrium points since

$$P_{4t}/P_{1t} = (P_{2t}/P_{1t})(P_{3t}/P_{2t})(P_{4t}/P_{3t}) \quad . \quad . \quad . \quad . \quad (41)$$

the terms on the right hand side of which follow from the equilibrium states. Using these values of P_{4t}/P_{1t} Fig. 8 enables us to find directly the value of $V/\sqrt{T_\alpha}$ consistent with our particular $Q\sqrt{T_{4t}}/P_{4t}$ by interpolation among the branches of the curve. Of course if the value of $Q\sqrt{T_{4t}}/P_{4t}$ for the equilibrium point exceeds the choking flow for the final nozzle then that point is impossible irrespective of the value of P_{4t}/P_{1t}. Or if we require, in order to meet a certain value of $Q\sqrt{T_{4t}}/P_{4t}$, a value of P_{4t}/P_{1t} in Fig. 8 greater than that corresponding to $V/\sqrt{T_\alpha}=0$, that point is also inadmissible.

In this way all the possible running points on the compressor characteristics may be assigned a value of $V/\sqrt{T_\alpha}$, or alternatively be proved to correspond to the choking flow of the final nozzle irrespective of $V/\sqrt{T_\alpha}$.

Since to every point on each of the compressor $N/\sqrt{T_{1t}}$ curves we can only assign one value of $Q\sqrt{T_{4t}}/P_{4t}$ consistent with equilibrium running, only one point, if any, on each said curve can correspond to the $Q\sqrt{T_{4t}}/P_{4t}$ for the nozzle choking flow. These single points for the various speeds may be connected and hence the final nozzle choking line may be transferred as a unique curve to the compressor characteristics. The lines of constant $V/\sqrt{T_\alpha}$ branch away to the left of it, the lower values of $V/\sqrt{T_\alpha}$ breaking off first (see Fig. 9). Thus we find that the choking part of the jet pipe curve lies roughly parallel to the compressor surge line. The non-choking branches bend in towards the surge line; but as $V/\sqrt{T_\alpha}$ increases the jet pipe flow at any given P_{4t}/P_{1t} (*i.e.* at any given P_{2t}/P_{1t}) increases and the running point moves farther away from the surge line.

4. PLOTTING THE EQUILIBRIUM RUNNING LINES.

We have seen that the most important variable linking the compressor and turbine when running in equilibrium is the parameter T_{3t}/T_{1t}. It may be useful here to set out the various steps involved in locating the equilibrium running point, given a value for this parameter. We do this in a table stating the known conditions at the beginning of each step on the left and the results obtained from these data on the right.

At this stage we check to see that the two values of $(T_{3t}-T_{4t})/T_{3t}$ are co-incident. If they are not we try another value of $Q\sqrt{T_{1t}}/P_{1t}$, on the same $N/\sqrt{T_{1t}}$ curve, and repeat until agreement is obtained. This process may be carried out to a high degree of accuracy if desired and when completed gives all the conditions necessary for the exact equilibrium of the compressor-turbine unit for the chosen speed $N/\sqrt{T_{1t}}$ and heat input parameter T_{3t}/T_{1t}.

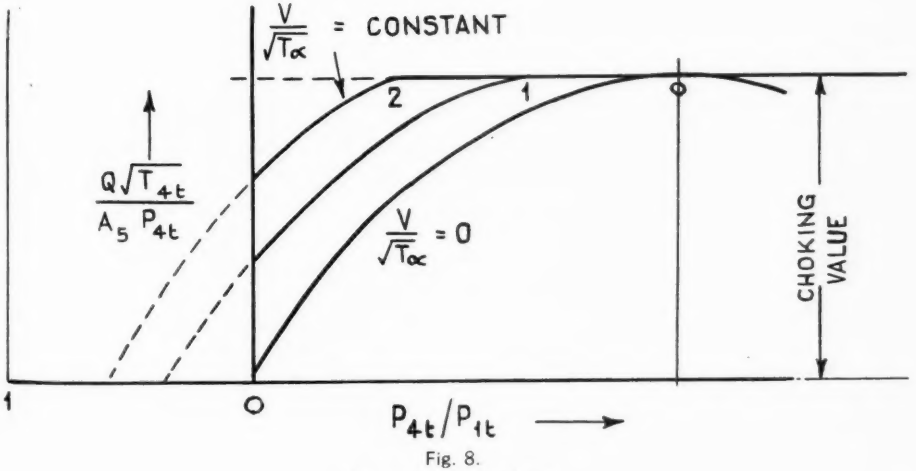

Fig. 8.
Jet pipe characteristics.

TABLE

Given $N/\sqrt{T_{1t}}$ and T_{3t}/T_{1t}

From	*Follows*
A *trial value* of $Q\sqrt{T_{1t}}/P_{1t}$, $N/\sqrt{T_{1t}}$, and the compressor characteristics	P_{2t}/P_{1t}, T_{2t}/T_{1t}
$Q\sqrt{T_{1t}}/P_{1t}$, P_{2t}/P_{1t} and T_{2t}/T_{1t}	$Q\sqrt{T_{2t}}/P_{2t}$
$Q\sqrt{T_{2t}}/P_{2t}$ and the combustion pressure loss curve (equation 15)	P_{3t}/P_{2t}
$Q\sqrt{T_{1t}}/P_{1t}$, P_{2t}/P_{1t}, P_{3t}/P_{2t} and T_{3t}/T_{1t}	$Q\sqrt{T_{3t}}/P_{3t}$
$N/\sqrt{T_{1t}}$ and T_{3t}/T_{1t}	$N/\sqrt{T_{3t}}$
$(T_{2t}-T_{1t})/T_{1t}$, T_{3t}/T_{1t} and the balance of work between compressor and turbine	$(T_{3t}-T_{4t})/T_{3t}$
$Q\sqrt{T_{3t}}/P_{3t}$, $N/\sqrt{T_{3t}}$ and the turbine characteristics	$(T_{3t}-T_{4t})/T_{3t}$

By taking other speed lines more equilibrium running points satisfying the given T_{3t}/T_{1t} may be found. Further sets of points may be located after altering the value of T_{3t}/T_{1t}. Ultimately by joining points of constant T_{3t}/T_{1t} a set of curves crossing the $N/\sqrt{T_{1t}}$ lines is obtained which may be regarded as the equilibrium running lines for the turbine-compressor combination (see Fig. 10). These equilibrium running lines will appear on the pressure ratio characteristics of the compressor as a series of nearly-parallel straight lines or lines of slight curvature, pointing towards the origin. If the turbine is choking,

$$Q\sqrt{T_{3t}}/P_{3t} = \text{a constant}$$

$$\therefore \quad (Q\sqrt{T_{1t}}/P_{1t})(P_{1t}/P_{3t})(\sqrt{T_{3t}/T_{1t}}) \text{ is constant,}$$

so that a line of constant T_{3t}/T_{1t} requires that

$$Q\sqrt{T_{1t}}/P_{1t} \propto (P_{3t}/P_{1t})$$

Now the difference between P_{3t}/P_{1t} and P_{2t}/P_{1t} is small so that with the turbine choking

$$(Q\sqrt{T_{1t}}/P_{1t}) \propto (P_{2t}/P_{1t})$$

very nearly. Thus lines of constant T_{3t}/T_{1t} become very nearly straight lines through the origin for all points on the choking part of the turbine characteristics.

We now complete the calculation of the non-dimensional quantities for the equilibrium point thus

From	*Follows*
$(T_{3t}-T_{4t})/T_{3t}$, $N/\sqrt{T_{3t}}$ and the turbine characteristics	P_{3t}/P_{4t}
$Q\sqrt{T_{3t}}/P_{3t}$, T_{3t}/T_{4t} and P_{3t}/P_{4t}	$Q\sqrt{T_{4t}}/P_{4t}$
(P_{2t}/P_{1t}), (P_{3t}/P_{2t}), (P_{4t}/P_{3t})	P_{4t}/P_{1t}
$Q\sqrt{T_{4t}}/P_{4t}$, P_{4t}/P_{1t} and the jet pipe characteristics	$V/\sqrt{T_a}$ (or see below)

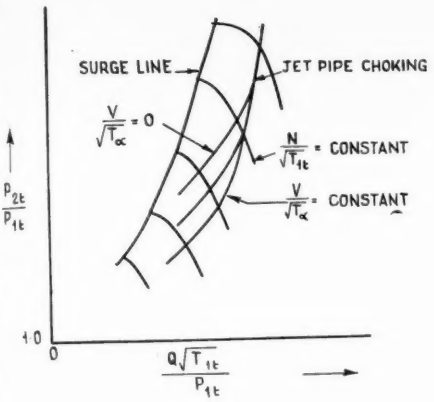

Fig. 9.
Jet pipe characteristics plotted on compressor characteristics.

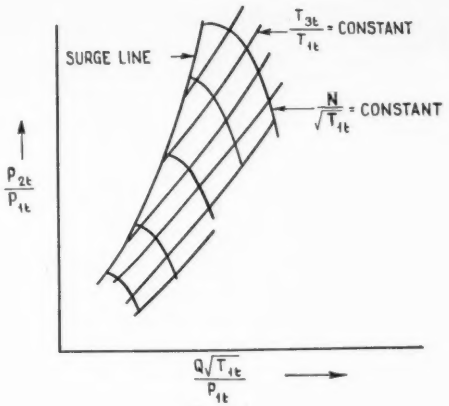

Fig. 10.
Lines of constant T_{3t}/T_{1t} plotted on compressor characteristics.

The last step gives the non-dimensional forward speed consistent with the particular equilibrium point only if it is a possible point *i.e.* V/T_a is greater than zero, and if the final nozzle is not choking. As explained in the third section if the $Q\sqrt{T_{4t}}/P_{4t}$ required for equilibrium exceeds the choking flow then the value of T_{3t}/T_{1t} needs increasing until the choking value is obtained. Such a point lies on the choking line of the final nozzle and cannot be associated with any particular value of $V/\sqrt{T_a}$.

The components of the engine can in general only be matched exactly at one equilibrium point called the design point. As the name implies this is the point where the engine variables such as the pressures, temperatures and mass flows will be those called for by the engine design data; and the intake, compressor, combustion chamber and turbine will all meet the required conditions simultaneously.

As we have seen, every change in running conditions requires a new "non-dimensional" equilibrium point. The pressures, temperatures and mass flows, and in particular the component efficiencies, will alter from point to point. To predict the performance of the engine when running away from the design condition it is necessary to locate each new equilibrium point.

5. LAWS GOVERNING THE ENGINE PERFORMANCE.

Having traced the jet pipe characteristics on the compressor characteristics we may fix the equilibrium running point for any flight condition defined by V, N, P_a and T_a, since every possible running point on the compressor characteristics is fixed by the two parameters $V/\sqrt{T_a}$ and $N/\sqrt{T_{1t}}$. The former can be calculated immediately. The latter follows from $N/\sqrt{T_a}$ and T_{1t}/T_a (Fig. 2).

All the other non-dimensional quantities of the engine follow these two parameters. Thus from the compressor diagram we have at once $Q\sqrt{T_{1t}}/P_{1t}$, $N/\sqrt{T_{1t}}$, P_{2t}/P_{1t}, T_{2t}/T_{1t} and T_{3t}/T_{1t} as single-valued functions of the pair of parameters. We denote these quantities by f_1, f_2, f_3, f_4 and f_5, where f refers to function $V/\sqrt{T_a}$, $N/\sqrt{T_a}$.

i.e.

$$Q\sqrt{T_{1t}}/P_{1t} = f_1\,(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (42)$$

$$N/\sqrt{T_{1t}} = f_2\,(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (43)$$

$$P_{2t}/P_{1t} = f_3\,(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (44)$$

$$T_{2t}/T_{1t} = f_4\,(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (45)$$

$$T_{3t}/T_{1t} = f_5\,(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (46)$$

From the intake curves we have P_{1t}/P_a and T_{1t}/T_a as functions of $V/\sqrt{T_a}$ and we may write

$$P_{1t}/P_a = f_6(V/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad . \quad . \quad (48)$$

$$T_{1t}/T_a = f_7(V/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad . \quad (49)$$

From the above we may deduce other quantities which must all be functions of the same basic parameters $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$.

Thus from (42), (48) and (49) we know

$$Q\sqrt{T_a}/P_a = f_8(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (50)$$

and from (42), (43) and (44)

$$Q\sqrt{T_{2t}}/P_{2t} = f_9(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (51)$$

From (51) and the combustion pressure loss characteristic (15) we have P_{3t}/P_{2t} which must therefore be another unique function of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$, *i.e.*

$$P_{3t}/P_{2t} = f_{10}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (52)$$

The equilibrium point may now be located on the turbine characteristics. To do this we require $N/\sqrt{T_{3t}}$ and $(T_{3t}-T_{4t})/T_{3t}$. The former follows from $N/\sqrt{T_{1t}}$ and T_{3t}/T_{1t} (equations (43) and (46)) and so is again a unique function of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$, *i.e.*

$$N/\sqrt{T_{3t}} = f_{11}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (53)$$

$(T_{3t}-T_{4t})/T_{3t}$ follows from T_{2t}/T_{1t} and T_{3t}/T_{1t} and the balance of work between the turbine and compressor (equation (19)) and so must be another function of the basic parameters.

$$\therefore \quad (T_{3t}-T_{4t})/T_{3t} = f_{12}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (54)$$

With the point located on the turbine characteristics we may read off P_{3t}/P_{4t} which must again be a unique function of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$.

$$\therefore \quad P_{3t}/P_{4t} = f_{13}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (55)$$

Now T_{1t}/T_a, T_{3t}/T_{1t} and $(T_{3t}-T_{4t})/T_{3t}$ (equations (49), (46) and (54)) may be used to calculate T_{4t}/T_a, hence T_{4t}/T_a is a unique function of the same basic parameters, *i.e.*

$$T_{4t}/T_a = f_{14}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (56)$$

In the same way P_{4t}/P_a follows from P_{1t}/P_a, P_{2t}/P_{1t}, P_{3t}/P_{2t} and P_{4t}/P_{3t} (equations (48), (44), (52) and (55)) and is another unique function of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$, *i.e.*

$$P_{4t}/P_a = f_{15}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (57)$$

Equation (25) shows that T_{4t}/T_a and P_{4t}/P_a give $V_j/\sqrt{T_a}$, the jet velocity in "non-dimensional" form. Hence $V_j/\sqrt{T_a}$ is a unique function of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$, *i.e.*

$$V_j/\sqrt{T_a} = f_{16}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (58)$$

The thrust developed by the engine is

$$F = Q(V_j - V)/g \quad . \quad . \quad . \quad . \quad . \quad . \quad (26)$$

which may be written

$$F/P_a = (Q\sqrt{T_a})/(gP_a)\{(V_j/\sqrt{T_a}) - (V/\sqrt{T_a})\} \quad . \quad . \quad . \quad (59)$$

and since the quantities on the right hand side of this equation, as shown by equations (50) and (58) are unique functions of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$ then it follows that F/P_a is a unique function of them also:

$$F/P_a = f_{17}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (60)$$

Thus for equilibrium running, the engine condition is defined by the parameters $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$, and F/P_a is a unique function of these parameters.

It is natural therefore to consider F/P_a as the "non-dimensional" thrust of the engine. As was pointed out in the introduction the true non-dimensional quantity would be F/D^2P_a where D is a representative linear dimension, but for any one engine this may be dropped from the expression.

Since we know $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$ for all equilibrium running points on the compressor diagram, it follows that F/P_a is known for all such points.

It also follows that engine test results of thrust measurement should be plotted in the form F/P_a against the two parameters $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$. It is usual to plot F/P_a against $N/\sqrt{T_a}$ as abscissca for a series of values of $V/\sqrt{T_a}$. The curve for $V/\sqrt{T_a}=0$ will correspond to the bench test of the engine (see Fig. 11).

Sometimes the parameters are quoted in the forms $(F\times 14.7)/P_a$, $N\sqrt{288}/\sqrt{T_a}$ and $V\sqrt{288}/\sqrt{T_a}$ so that the scales show at a glance the actual thrusts and speeds at sea level.

For the fuel consumption, from equations (8) and (9) we have

$$\eta_c H Q_F = C_{P2.3}.\, Q\,(T_{3t} - T_{2t})$$

which, treating the combustion efficiency η_c and specific heat $C_{P2.3}$ as constants we may write

$$HQ_F = \text{constant}\; Q\,(T_{3t} - T_{2t}) \quad . \quad . \quad . \quad . \quad . \quad (61)$$

The last equation may be rearranged into the following "non-dimensional" form on dividing through by $P_a\sqrt{T_a}$.

$$(HQ_F)/(P_a\sqrt{T_a}) = \text{constant}\;(Q\sqrt{T_a}/P_a)\,(T_{3t} - T_{2t})/T_a \quad . \quad . \quad (62)$$

The right hand side of (62) may be obtained from $Q\sqrt{T_a}/P_a$, T_{3t}/T_{1t}, T_{1t}/T_a and T_{2t}/T_{1t}, *i.e.* from equations (50), (46), (49) and (45) respectively and all these quantities are unique functions of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$. Hence we may write $HQ_F/P_a\sqrt{T_a}$ as a unique function of the same, *i.e.*

$$HQ_F/P_a\sqrt{T_a} = f_{18}\,(V/\sqrt{T_a},\; N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (63)$$

As mentioned in the first section of the paper $HQ_F/P_a\sqrt{T_a}$ is not truly "non-dimensional" but needs dividing by D^2 to make it so. For any given engine the representative linear dimension D is common, and the heating value H is usually a constant. So H and D are normally left out and $Q_F/P_a\sqrt{T_a}$ regarded as the "non-dimensional" fuel consumption. We have shown that for equilibrium running, it is a parameter which depends only on $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$. Therefore, for steady conditions, fuel consumption measurements should be plotted as curves of $Q_F/P_a\sqrt{T_a}$ against $N/\sqrt{T_a}$ and we expect to get a unique curve for each value of $V/\sqrt{T_a}$ (see Fig. 12).

In using the fundamental laws

$$F/P_a = f_{17}(V/\sqrt{T_a},\; N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (60)$$

$$Q/(P_a\sqrt{T_a}) = f_{18}(V/\sqrt{T_a},\; N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (63)$$

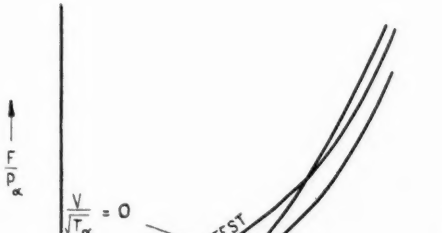

Fig. 11.
Plot of "non-dimensional" thrust for turbo-jet engine.

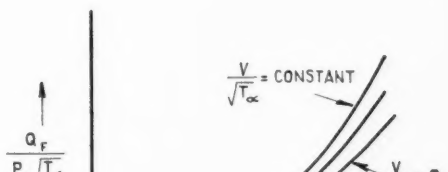

Fig. 12.
Plot of "non-dimensional" fuel consumption for turbo-jet engine.

it should be pointed out in the former we imply, in using equations (48) and (25), that the intake and jet pipe efficiencies are unique functions of $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$; and that in the latter we imply a constant specific heat $C_{P2.3}$ and combustion efficiency η_c. Any error in these assumptions will show up as scatter in plotted results. We can say however that only fairly severe charges in intake jet pipe, or combustion efficiency will vitiate these laws; and that in such cases the scatter of test points will indicate that all is not as it should be with the engine.

6. OPERATION AT CONSTANT R.P.M.

From the stressing point of view it is important to consider the behaviour of the engine when running at maximum r.p.m. over the altitude range.

At constant r.p.m. with increasing altitude $N/\sqrt{T_{1t}}$ increases as the atmospheric temperature falls.

If the propelling nozzle is choking the operating point on the compressor characteristics then moves upwards along the nozzle choking line.

Along this line $Q\sqrt{T_{4t}}/P_{4t}$ = a constant X (say), *i.e.*

$$(Q\sqrt{T_{3t}}/P_{3t})(P_{3t}/P_{4t})(\sqrt{T_{4t}}/T_{3t}) = X \quad . \quad . \quad . \quad (64)$$

These are the parameters governing the turbine.

Now for a steady increase in altitude at constant r.p.m. the turbine operating point will stay on a uniform curve first because all the $N\sqrt{T_{3t}}$ lines approach one another at the choking condition of the turbine, and secondly because $N/\sqrt{T_{3t}}$ tends to remain constant. On such a uniform curve there is only one value for $Q\sqrt{T_{3t}}/P_{3t}$, P_{3t}/P_{4t} and T_{4t}/T_{3t}. Further as $Q\sqrt{T_{3t}}/P_{3t}$ increases, P_{3t}/P_{4t} increases at a greater rate than $\sqrt{T_{4t}}/T_{3t}$ decreases. Hence there is only one point where equation (64) can be satisfied, and therefore the turbine must operate at a fixed non-dimensional point, *i.e.*

$$(T_{3t} - T_{4t})/T_{3t} = \text{a constant.}$$

But from the balance of turbine and compressor work (equation 19)

$$(T_{3t} - T_{4t}) \propto (T_{2t} - T_{1t})$$

Therefore, so long as the propelling nozzle is choking

$$(T_{2t} - T_{1t}) \propto T_{3t} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (65)$$

Now the term $(T_{2t} - T_{1t})$ represents the compressor work per lb. of air mass flow and may be written

$$(T_{2t} - T_{1t}) = fN^2 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (66)$$

where f the "work factor" is practically constant over the operating range but decreases as a rule due to compressibility losses at the highest values of $N/\sqrt{T_{1t}}$.

From equations (65) and (66) it follows that with the propelling nozzle choking

$$fN^2/T_{3t} = \text{a constant} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (67)$$

So we deduce that at constant r.p.m. T_{3t} is a constant, except at high values of $N/\sqrt{T_{1t}}$ where it will decrease in proportion to the fall-off in the work factor of the compressor.

This is a valuable property of the turbo-jet. The designer naturally wishes to claim the utmost from the engine in flight but must limit the r.p.m. and maximum gas temperature irrespective of the altitude. We see that with a choking propelling nozzle operation to the maximum r.p.m. will automatically demand the maximum gas temperature; though if there is a loss of work factor at the highest altitudes the gas temperature will not reach its permissible maximum and in consequence there will be some loss of thrust.

If the propelling nozzle is not choking, then by following a line of constant $V/\sqrt{T_a}$ on the compressor characteristics (Fig. 9) it is apparent that as $N/\sqrt{T_{1t}}$ increases $Q\sqrt{T_{1t}}/P_{1t}$ increases at a greater rate than when the propelling nozzle is choking to correspond with the increase in $Q\sqrt{T_{4t}}/P_{4t}$. This requires the X of equation (64) to increase with increase of altitude instead of remaining constant. It follows that $(T_{3t} - T_{4t})/T_{3t}$ must also increase, *i.e.* $(T_{2t} - T_{1t})/T_{3t}$ increases as the altitude increases or fN^2/T_{3t} increases.

In this case with a constant r.p.m. as altitude increases, T_{3t} must decrease on two counts, first to allow for any decrease in f and secondly to allow for the increase in fN^2/T_{3t}. The corresponding loss in thrust will therefore be greater than when the propelling nozzle is choking.

7. LINK WITH THE AIRCRAFT PARAMETERS.

The drag of the aircraft at a given equivalent air speed V_i ft. per second is given by the sum of the profile and induced drags, and is expressed by the equation

$$D = D_{o_{100}}(V_i/100)^2 + I_{100}(V_i/100)^{-2} \quad . \quad . \quad . \quad . \quad . \quad (68)$$

where D_{o100} and I_{100} represent the profile and induced drags at a speed of 100 ft. per second.

In steady level flight the drag D equals the thrust F. Then the "non-dimensional" thrust F/P_a is given by dividing equation (68) through by P_a, *i.e.*

$$F/P_a = (D_{o_{100}}/P_a)(V_i/100)^2 + (I_{100}/P_a)(V_i/100)^{-2} \quad . \quad . \quad . \quad (69)$$

Now
$$V_i^2 = \rho V^2/\rho_o \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (70)$$

where ρ_o is the standard sea level density, and ρ is the atmospheric density at the altitude in question which is given by P_a/RT_a. Substituting for ρ in equation (70) gives

$$V_i^2 = (P_a/\rho_o R)(V/\sqrt{T_a})^2 \quad . \quad . \quad . \quad . \quad . \quad (71)$$

We also have:

$$D_{o_{100}} = (\rho_o/2g)(100^2 C_{D_o} S) \quad . \quad . \quad . \quad . \quad . \quad (72)$$

where C_{D_o} is the basic profile drag coefficient and S the wing area, and

$$I_{100} = 2gKW^2/(\pi\rho_o A 100^2 S) \quad . \quad . \quad . \quad . \quad . \quad (73)$$

where K is a constant approximately 1.2, W is the aircraft weight and A is the aspect ratio.

Substituting for V_i, $D_{o_{100}}$ and I_{100} from (71), (72) and (73) into (69) gives

$$F/P_a = a(V/\sqrt{T_a})^2 + b(W/P_a)^2(V\sqrt{T_a})^{-2} \quad . \quad . \quad . \quad . \quad (74)$$

where $a = C_{D_o} S/2gR$ and $b = 2gKR/\pi AS$, which are both constants for a particular aircraft. Thus the "non-dimensional" thrust F/P_a is a unique function of $V/\sqrt{T_a}$ and W/P_a.

If the aircraft is in a steady climb of vertical rate C ft. per second the power equation is

$$F.V. = D.V. + WC$$

or
$$F = D + WC/V$$

$$= D + (WC\sqrt{\sigma}/100)(100/V_i) \quad . \quad . \quad . \quad . \quad . \quad (75)$$

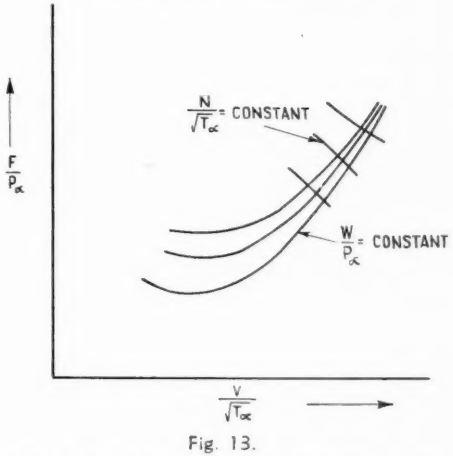

Fig. 13.

Level flight performance plot for turbo-jet aircraft.

Substituting for V_i (equation (71)) and putting $\sigma = P_a/RT_a\rho_0$ gives

$$F/P_a = D/P_a + (W/P_a)(C/\sqrt{T_a})(V/\sqrt{T_a})^{-1} \quad . \quad . \quad . \quad . \quad (76)$$

For D/P_a we may put the F/P_a for level flight given by equation (74). This gives the general equation

$$F/P_a = a(V/\sqrt{T_a})^2 + b(W/P_a)^2(V/\sqrt{T_a})^{-2} + (W/P_a)(C/\sqrt{T_a})(V/\sqrt{T_a})^{-1} \quad . \quad (77)$$

which shows the relationship between the "non-dimensional" thrust (F/P_a), non-dimensional forward speed ($V/\sqrt{T_a}$), non-dimensional rate of climb ($C/\sqrt{T_a}$) and "non-dimensional" all-up weight (W/P_a) of the aircraft.

Finally let us consider the fuel consumption. We have equation (63) namely:

$$HQ_F/P_a\sqrt{T_a} = f_{18}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (63)$$

which shows that for equilibrium running the "non-dimensional" fuel consumption $Q_F/P_a\sqrt{T_a}$ depends only on $V/\sqrt{T_a}$ and $N/\sqrt{T_a}$.

Equation (60) shows that the "non-dimensional" thrust is a unique function of the same parameters

$$F/P_a = f_{17}(V/\sqrt{T_a},\ N/\sqrt{T_a}) \quad . \quad . \quad . \quad . \quad (60)$$

We are primarily interested in the fuel consumption for level flight, and here equation (74) shows that F/P_a is a simple function of $V/\sqrt{T_a}$ and W/P_a so that we may write

$$F/P_a = f_{19}(V/\sqrt{T_a},\ W/\sqrt{P_a}) \quad . \quad . \quad . \quad . \quad (78)$$

By comparing equations (60) and (63) it is seen that a certain value of F/P_a will be associated with a certain value of $HQ_F/P_a\sqrt{T_a}$ for every engine operating point. Equation (78) shows that F/P_a is known given the $V/\sqrt{T_a}$ and W/P_a of the aircraft. Therefore $HQ_F/P_a\sqrt{T_a}$ will depend only on the $V/\sqrt{T_a}$ and W/P_a of the aircraft.

It is usual to plot the level speed performance of a turbo-jet propelled aeroplane as curves of F/P_a against $V/\sqrt{T_a}$ for constant values of W/P_a. Unique lines of constant $N/\sqrt{T_a}$ may be plotted across the W/P_a lines. In this way we get a network of curves covering four of the main parameters (Fig. 13). Lines of constant $Q_F/P_a\sqrt{T_a}$ may be plotted on the same graph. Alternatively the latter are given separately on a base of $N/\sqrt{T_a}$ for various values of $V/\sqrt{T_a}$ (Fig. 12).

8. CONCLUSION.

We have attempted in this paper to show the application of certain basic "non-dimensional" parameters in the performance analysis of the turbo-jet. Once the principles have been grasped these "non-dimensional" parameters are of immense value in elucidating the behaviour of the engine at operating points away from the design point. They show the co-ordinates on which the thrust and fuel consumption of the engine should be plotted and the link with the fundamental aircraft parameters.

The turbo-jet is a classic type of aircraft propulsive unit. The simplicity of its fundamental laws when expressed in "non-dimensional" form is in keeping with its mechanical simplicity. The neat form of its basic parameters brings into use a novel "non-dimensional" rearrangement of the aircraft drag equation which links naturally with the engine laws.

It has been shown how the so-called basic parameters of the turbo-jet are derived from the characteristics of the various components of the engine, and how it is possible to predict the performance of the engine from these characteristics. Investigations into the effect of changes in component characteristics take the form of problems in the geometry of these curves.

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THE ROYAL AERONAUTICAL SOCIETY

FLIGHT IN NATURE AND IN HUMAN SOCIETY

by

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A STUDY of the evolution of the flight faculty—and in particular its effects in the realm of Nature and in human affairs—reveals analogies which are, perhaps, of more than technical interest. When due proportions have been established, it will be seen that the current period in the development of human locomotion is comparable to that very remote epoch during which Nature, having herself solved the problem of flight, exercised the faculty in accord with some inscrutable law. The analogy gains strength from the fact that changes of a profound kind in the conditions of life on this planet followed the evolution of flight in Nature; and that changes no less profound in character have taken, and are now taking, place as the result of man's so-called "conquest of the air" brought about by technological advancement.

Flight condition favourable to survival.

Man achieved flight late in his history; but Nature, also, realised locomotion in the air only in a second stage of the evolution of living forms, the development of wings being the effect of successive adaptation. How widespread is the power of flight may be judged from some calculations cited by the zoologist, Zschokke,[1] that among 420,000 animal species existing at present at least 260,000 are able to move through the air; namely, of the species living to-day some 62 per cent. is adapted in varying degrees to aerial locomotion. Moreover, it is suggested that if the creatures whose habitat is water are excluded from these calculations, the ratio of flying to non-flying species rises to 75 per cent. The same authority estimates that among 280,000 species of insects, 240,000 are capable of flight; and to this number of flyers 13,000 species of birds and 600 of bats would have to be added. It is fair to deduce from these figures—without undue regard for their exactness—that the unconstrained motion of flight offered, generally, very favourable conditions for survival. This is the first of several conclusions to be drawn from a study of the natural phenomenon.

In Nature the flight faculty was used to escape from preying creatures, to facilitate observation, and to reach and collect food more easily. The flyers were able to alight on the islands arising from the sea and in the valleys forming on glaciers; pollination and the spreading of life was thereby assisted —the flying insects became the pioneers of life and conservers of the vegetable kingdom. The variety and charm of flowers was largely established by the phenomenon of insect pollination which is necessary, not only for the conservation of a great part of the vegetable world, but also for its beauty. By producing changes such as these, the faculty of flight may be said to have given rise to a new environment, and it would require an effort in imagination to conceive what the conditions on earth would have been to-day had the faculty not developed in Nature.

Evidence of human aspiration.

As already observed, man achieved flight late in his history. Lord Brabazon[2] and others have suggested that gliding and soaring (motorless) flight came very much later

than was possible; that there is nothing in the construction of a modern high-efficiency glider which could not have been effected a thousand years ago and more, had the desire existed and the basic principles been understood. Unlike the motor-car, for example, which necessitated machine tools, metallurgical and engineering ability, the building of a glider required only a moderate degree of craftsmanship and materials which have been available from the earliest times. In this connection, it is of interest to note that diametrically opposed views concerning the alleged constant aspiration of humanity to fly have been advanced from time to time. For instance, the anthropologist, Berthold Laufer,[3] held the opinion that the desire to fly is as old as man himself, and his contention was reinforced by wide scholarship in the art and literature of the orient.

Conversely, Professor R. Giacomelli[4] has suggested from the standpoint of engineering science and of biology, that the idea of achieving flight did not attract serious attention until modern times, although not on account of any inherent difficulty in the problem. Evidence of a desire to emulate Nature is certainly contained in the works of art of the earliest civilisations and in the traditions and legends of many peoples; but, until the beginning of the 16th century, it appears to have had only the force of vague aspiration.

Some historians in this subject have argued, in support of the legends, that the imaginative faculty of the human mind is unable to conceive things which have absolutely no reality in existence, and it is because primitive man was unable to render an intelligent account of the practices of his day that such records as exist are unconvincing. But it is plain that most of the alleged flights, as represented in mythology and legend, are physically impossible; and it is probable that it was merely the fact of a declared intention, or of an attempt, which provided the foundation for the story.

Application of natural laws to artificial flight.

It is in man's application of natural laws and principles to flight that the greater interest lies. In Nature, the evolution of true flight was a process probably extending over millions of years and consisting of innumerable stages and gradations within stages whereby some developments were arrested, some started late, and some were abandoned. The total of our knowledge of these processes indicates the ability to glide, and possibly to soar, was almost certainly a preliminary stage in the development of true flight, therefore it is not surprising perhaps to find that all attempts to achieve full-scale mechanical flight without *first* passing through the transitional stage, found to be obligatory in Nature, failed.

Langley's abortive man-carrying experiments and the Wright brothers' successful power flights on 17th December 1903 following their investigation of gliding flight—well illustrate this fact.[5] Moreover, when it is remembered that the solution attempted by the method of buoyancy in the balloon and airship — of which Nature presents no examples—was not wholly satisfactory, the analogy between natural and artificial flight is strengthened.

In Nature, all sustentation in flight is based on the dynamic principle, namely on the existence of a relative velocity between the air and the supporting surfaces, the statical principle (buoyancy) being confined to the locomotion of creatures in water. It is significant perhaps that man's application of the former principle to sustentation in water with the hydroplane (another device having no natural counterpart) has not advanced far and it would seem that Nature was guided by factors of suitability and efficiency. Mechanical flight is the application of fundamental principles used in Nature (not necessarily the best application, although man's machines are probably more efficient) with the employment of heat engines on an extravagant scale

analogous to the rapid digestion of a bird—"The always hungry stomach of this warm-blooded master of motion."

FUNDAMENTAL SIGNIFICANCE OF FLIGHT.

The significance of flight lay in the fact that it constituted the development of three-dimensional movement in the medium which was all-pervading with regard to the earth's surface—the air. Hitherto, animal movement had been restricted to the land and water; and although mobility in these media may also have been three-dimensional, the air, which on account of its lower density and comparatively unobstructed character was most suitable for *rapid* motion, had not been exploited.

There can be no answer to the question why the faculty of flight developed in Nature, other than the common generalisation that it was evolved to meet and give rise to a new environment, as evidenced by the uses already described. But it is plain that man, having acquired the same faculty by his technical skill, immediately applied the art in his own struggle for existence; to him it was activity wholly in a new element and therefore revolutionary by implication as well as by direct effect. It made possible, among other things, the practice of total war; its inception (coincident with other technical developments) marked the beginning of a new era in the history of the world. In its relation to human society, speed in physical communication is the fundamental factor.

SOCIAL CHANGES RESULTING FROM AVIATION.

The first, and hitherto most pronounced, effect of the introduction of artificial flight is its influence on the method and scope of warfare. Used as a weapon of offence, the aeroplane with its load of explosive is a force which can directly and speedily reach the populations at war and the means of waging war. The advent of the aeroplane made possible the conception and realisation of total war. It was early foreseen that the enhanced mobility with much increased speed which flight afforded, would produce radical changes in man's behaviour and effects in the reorganisation of his society; but the full extent of these changes was not, it seems, understood. The operation of war was transformed from one of a comparatively civilised kind to one of savage barbarism affecting the bulk of society and the wholesale destruction of industry produced repercussions of the gravest kind. Certainly the Hiroshima episode can be regarded as initiating a change in ethical standards made feasible by the development of flight; for whatever the lethal element may be, it must be transported or impelled through the air and the factor of its locomotion is the factor which renders it so destructive.

More significant perhaps than the military application is the conception (now partly realised) of air transport on a "global basis," which ignores obstacles placed by Nature on surface travel and is concerned only with the shortest distance between any two places on the earth's surface. Such facility of communication bestows on those who control it exceptional opportunities for the dissemination of their particular culture and for the social and economic development of large tracts of country where no other or inadequate transport exists. Speed, by flight, may be an aid to the organisation and exploitation of the world's resources.

The history of transport shows that one important result of enhanced mobility was the tendency to form larger social and political groups. It is probable that this process will be accelerated by the development of air transport on a global basis; but whether such movement will assist in the establishment of a central sovereign authority for the whole human race is beyond immediate consideration. The project known as "space flight"—travel outside the atmosphere and leading to inter-planetary communication—is beyond practical consideration in its possible effect on human affairs.

In the final analysis, it would seem that man's use of the faculty of flight—and therefore the effects on his civilisation—will be determined by whether a regression to T. H. Huxley's so-termed, "cosmic process" in evolution takes place, or whether the opposing ethical process prevails. In the latter case, the benefits which humanity may derive from flight are likely to exceed even those enjoyed by Nature.

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REVIEWS

Wing Tips. The Identification of Birds in Flight. Roland Green. A. & C. Black, London, 1947. 6s. net. 64 pp.

In these days of nationalisation it is sometimes worth while reflecting on what has been done by private enterprise in the air. The author of this beautifully illustrated book enables the reflection to be made without any recriminations. Those who believe that nationalisation is the end-all and be-all of civil aviation will draw some comfort from the fact that since birds are not subsidised they are not expected to show a profit—unless they are off the ration.

This is a book intended for the bird lover by one who is not only one, but a first-class artist as well. His fifty-six drawings of birds in flight are delightful. To them are added explanatory drawings of the structure of a bird's wing, and a clearly written text of the various types of wing and flight manœuvres.

Altogether a book which should appeal to all those who love flying for flying's sake whether carried out by the dictatorial golden eagle or the national pigeon, both of which are so ably illustrated by the author.

Airliners and Airways of Today. S. E. Veale. Pilot Press Ltd., London, 1947. 21s. net. 328 pp.

In reviewing a book the present reviewer usually reads the author's preface first, for in the preface an author makes some statement of the scope of his book. Mr. Veale begins in a comprehensive way: " This book presents and discusses some of the problems that hinge upon the design and construction of airliners and their operation in public service."

There you are, ladies and gentlemen. Nothing could be fairer than that, could it? And, he adds, the book " can be read with understanding by those with no previous knowledge of the subject."

Judging by the comments made in Parliament on civil aviation, the diatribes in the daily press, and the remarks overheard in tubes, buses and trains, it is clear that most of Carlyle's population of England would profit by reading Mr. Veale's book. It is a good and comprehensive book covering aircraft and airways and airports and it is to be hoped that Mr. Veale will bring out new and up-to-date editions as often as the paper situation allows.

The fourth sentence of Mr. Veale's book should be printed in italics, for that sentence, of itself, gives the full explanation of the difficulties facing the country in air transport. *"Most of the material of the engineering sections,"* writes the author, *"is of British origin; that dealing with the operation of airliners comes chiefly from the United States."*

The reviewer has little doubt that in some future edition of Mr. Veale's book, he will be able to draw a considerable amount of his operational facts from British sources as well as his engineering. Flair for the air is in the blood of the British and they will ultimately take that place in the air which has been held for so long on the sea by the Merchant Navy.

Light Metals in Structural Engineering. L. Dudley. Temple Press Ltd., London, 1947. 30s. net. 216 pp.

The first ten chapters of Mr. Dudley's book—129 pages out of the total of 216—are largely applicable to all metals light or otherwise. The title is, indeed, somewhat misleading, as only one full chapter and part of another are fundamentally devoted to light metals.

Because of this, however, the book has a wider range of usefulness than the title would indicate. Those in aeronautics are inclined to look upon the light metals and their alloys as finding their chief use in aircraft construction from a structural point of view. *Light Metals in Structural Engineering* will be of value to all those who wish for an elementary practical introduction to the subject of strength of materials. Much of the value of this introduction is due to the many examples which are given throughout the text. It is the proper application of theory to practice which makes for the success of the final structure. There are many who have not the full knowledge nor indeed the ability to relate theory to practice in the best way, but there are many who have that deep desire to be as good at their craft as is possible within the range of their ability.

This book is for those engineers who have received a training limited by their environment but not limited by their desire for further knowledge.

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It is one of the great tragedies of the world that the rising costs of education, especially technical education, are making it more and more difficult to provide that education in the full measure which is necessary for proper appreciation of the task which lies ahead.

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In 1939 such a book as this would have cost 7s. 6d. to 10s. 6d. It now costs 30s.

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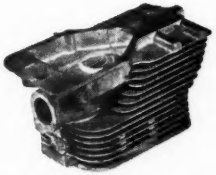

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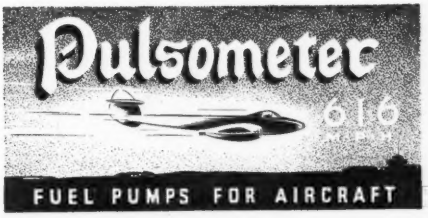

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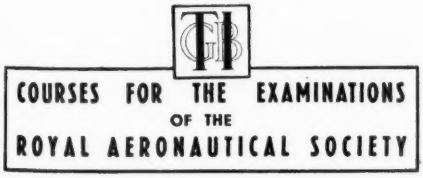

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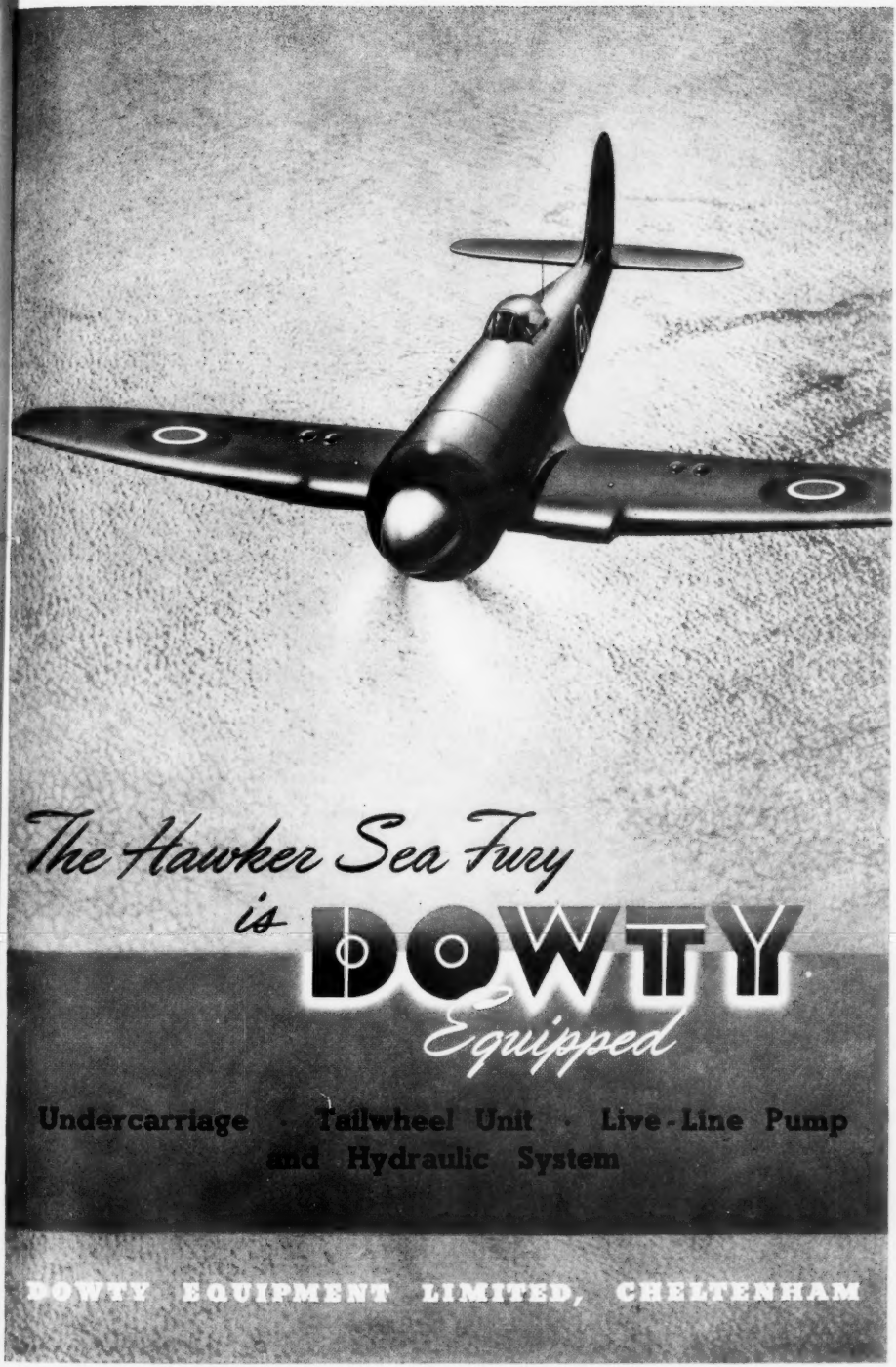

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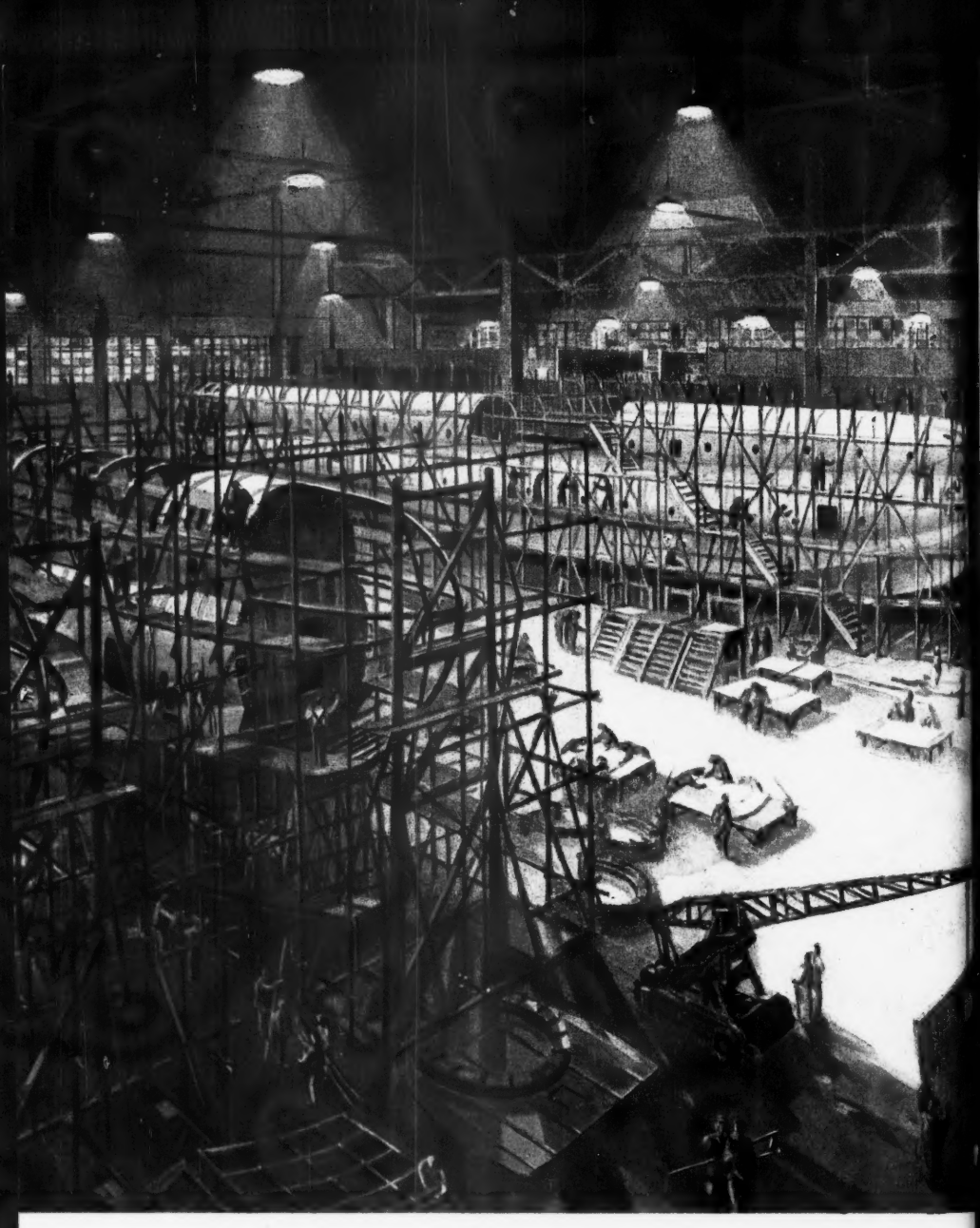

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